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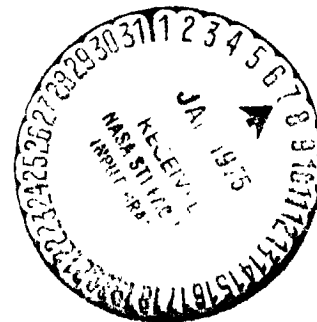
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MECHANICS OF FLIGHT TO DISTANT PLANETS

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INTRODUCTION

This monograph deals with the problems of flight to the distant /3* planets of the Solar System — Jupiter, Saturn, Uranus, Neptune, Pluto, as well as their natural satellites. The first four of these planets are often called planets of the Jupiter group, in distinction from the four planets of the Earth group — Mercury, Venus, Earth, and Mars. Planets of either one of these groups have certain common physical characteristics, which differ a good deal from those of planets of the other group.

For example, planets of the Earth group, in comparison with planets of the Jupiter group, are small, of low mass, relatively large average density, quite different chemical composition, well evident solid surfaces, and relatively little atmosphere. And although the surfaces of Mercury, Venus, Earth, and Mars are very dissimilar, they have one common property: a spacecraft sent from Earth can land there. Although the atmospheres of three of the planets are very different (the atmosphere of Mercury, if it exists at all, is so rarified that it is negligible), for all of them a surface landing and a lift-off are possible. On the other hand, a spacecraft reaching a planet of the Jupiter group may, at best, remain floating in the upper layers of the planet's gaseous envelope, and cannot lift off from the surface, since to do this it must climb through thousands of kilometers of atmosphere, subject to

*Numbers in the margin indicate pagination of original foreign text.

colossal pressures, high temperatures, and disastrous chemical reactions. But even if it could somehow survive the great depth of the atmosphere, subject to increasingly hot (to thousands of degrees) and dense layers, we have no idea how things would be at depth (and, incidentally, what is the depth?): an ocean of hydrogen and helium (and what depths?), or a solid surface (possibly consisting, for example, of crystalline metallic hydrogen on Jupiter). /4

Even if we find some reliable answers to these questions, it will probably take a very long time. We recall that there is as yet no reliable theory as to the internal structure, even of Earth. As regards Jupiter and similar planets, we have no reliable astrophysical data which could be the basis for calculations, nor accepted theories for physical laws obeyed by matter under the unusual conditions prevailing in the cores of planets.

The ninth planet of the solar system, Pluto, is not like the planets of the Jupiter group. We have no reliable information even on Pluto's size and mass, but it is clear from its appearance (and even from its period of rotation around its axis) that it should belong to planets of the Earth group. Of course, the conditions on the surface of this planet, which receives 1600 times less heat from the Sun than does Earth, cannot be remotely similar even to Mars and even more so to Venus or Mercury, but at least one cannot doubt the existence of a solid surface for Pluto. We could at least land on it. Apart from this, in all other respects of interest to astronauts, Pluto must be related to planets of the Jupiter group.

The outer region of the Solar System, occupied by the orbits of the planets of the Jupiter group, is a region of colossal distances and very great flight times (many years and even tens of years). This fact presents space technology with complex problems, different from those which must be solved in flights to Mars and Venus; one must remember that, beyond the orbit of Jupiter, the intensity of solar radiation is weaker by a factor of 25 than in the region of the Earth, and that it falls off in proportion to the square of the distance. Therefore, the solar energy cannot be used to drive the

equipment nor the radio transmitters of the spacecraft, and can be used for solar-electric propulsion only to a limited extent, if the spacecraft does not go too far from the inner regions of the Solar System. The solar cells, which have been so useful in flights to the Moon, Venus, and Mars, must be replaced by nuclear energy sources. A portable nuclear-electric source is an essential element in constructing an interplanetary spaceship. /5

However, our task here is not to consider the construction, the scientific equipment, nor the control systems of interplanetary spaceships. We will deal exclusively with the choice of ways and means to accomplish flights to Jupiter, Saturn, Uranus, Neptune, and Pluto, and we will explore the question of the energy required for a space flight or, which is the same thing, the required initial mass of rocket fuel, launched from Earth, or from a space rocket in orbit around the Earth.

We will consider a program for exploration of the planets only briefly, to the extent to which it impacts on the planning of interplanetary flight. As regards the planets themselves, we shall be concerned only with the data which one must have in examining space missions.

This monograph does not deal with planets lying beyond the orbit of Mars and the asteroid belt, nor with flight to these.

However, before passing on at once to flights to planets which interest us, we will briefly present the basic facts concerning space flight so that the later sections will be understood by the reader without reference to other books.

CHAPTER 1

GENERAL CONCEPTS IN THE THEORY OF INTERPLANETARY FLIGHT

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Flights Using a Simplified Model of Planetary Orbits

To a first approximation (but a very good approximation), the motion of all the planets and comets, and also the free (no engines) motion of spacecraft and space probes, i.e., artificial heavenly bodies, can be regarded as subject only to the acceleration of the Sun, if one can assume that the spacecraft is already far enough from the Earth and has not yet approached the planet (we shall see below how far from the Earth and how close to the planet). Hence it follows, from the law of universal gravitation, that spacecraft move relative to the Sun in an ellipse (a circle, in the special case) a parabola or a hyperbola. Any deviations from these relatively simple motions, caused by small forces, are called *perturbations*.

Motion in a parabola occurs when a spacecraft has a parabolic velocity, given by the formula

$$v_p = \sqrt{\frac{2\mu}{R}} \quad \text{or} \quad v_p = \sqrt{\frac{2K}{R}};$$

where $K = fM$ is the gravitational parameter of the Sun, f is the constant of universal gravitation, M is the mass of the Sun, and R is the distance from the Sun.

If the speed of a spacecraft is less than parabolic, it moves around the Sun as an artificial planet in an elliptical orbit. All the natural planets move in this way, and for most of them, the elliptical orbits are close to circular. The speed of motion in a circular orbit is

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$$V_{\text{circ}} = \sqrt{\frac{fM}{R}} = \sqrt{\frac{K}{R}}$$

and is called *circular*. The vectorial circular velocity must be perpendicular to the radius from the center of the Sun to the point occupied by the spacecraft.

It is not difficult to see that $V_p = V_{\text{circ}}\sqrt{2}$. Therefore, the parabolic speed relative to the Sun near a natural planet is approximately 41% larger than the speed of the planet itself.

If the spacecraft speed is greater than parabolic, it moves in a hyperbolic trajectory. Its speed is then called *hyperbolic*.

Moving along a parabola or a hyperbola, a spacecraft leaves the solar system; in the parabolic case, its speed — which diminishes steadily — tends to zero (as can be seen from the parabolic speed formula, letting $R \rightarrow \infty$) and in the case of a hyperbola, the speed decreases and tends to a certain value, the *residual speed at infinity* V_∞ . We recall the simple formula:

$$V_{\text{hyp}}^2 = \frac{2K}{R} + V_\infty^2 \quad \text{or} \quad V_{\text{hyp}}^2 = V_p^2 + V_\infty^2.$$

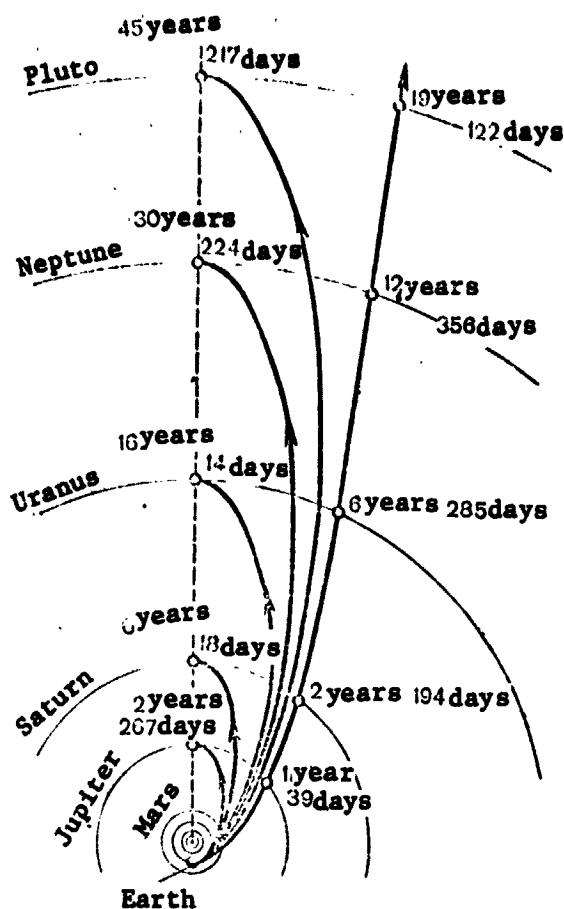


Figure 1. Flight trajectories: parabolic and cotangential ellipses. The flight duration is indicated.

At the mean distance of Earth from the Sun, $R_E = 149,600,000$ km, the circular speed is 29.785 km/sec. This is called the mean orbital speed of the Earth. The parabolic speed at the same distance is $29.785\sqrt{2} = 42.122$ km/sec.

If our planet had no gravitational attraction, it would be sufficient to add a speed of 12.337 km/sec in the direction of the Earth's motion to the speed of Earth, 29.785 km/sec, in order to obtain an initial velocity of 42.122 km/sec, which would give a flight in a parabola tangent to Earth orbit (Figure 1).

Using the simplified model of the Solar System, we assume that the orbits of all the planets are circular and lie in the plane of Earth orbit. It is then easy to show that, moving in a parabola, our spacecraft intersects in succession the orbits of Mars, Jupiter, Saturn, Uranus, Neptune, and Pluto. /8

If we leave the Earth with a somewhat smaller, but still large, speed, the motion will be along an elongated ellipse and will intersect all the orbits, but in order to reach a specified planet, one must choose an elliptic trajectory which does not intersect, but is only tangent to, the orbit of the planet. If this orbit was previously tangent to the Earth's orbit, it is a *cotangential* ellipse (i.e., an ellipse tangent to two concentric circles) and is the flight trajectory with minimum initial velocity (Figure 1), i.e., /9

it is the best, optimal from the energy viewpoint. Cotangential elliptical trajectories are also called Hohmann, from the German scientist V. Hohmann, who described these in a paper published in 1925.

Although we do not consider flights to the inner planets, Mercury and Venus, in this monograph, for the sake of completeness we should mention that the velocity increments (the minimal increments, in particular) to reach these must be directed opposite to the motion of the Earth. Therefore, Table 1 shows the values with a minus sign.

A flight along a Hohmann trajectory has a duration of one-half period of revolution in the complete cotangential ellipse. The formula for calculating the duration of a Hohmann transfer is:

$$T_{\text{Hohm}} = 64.5664 \sqrt{(1 + R_{\text{pl}})^3} \text{ days.}$$

Here the radius R_{pl} of the planetary orbit is expressed in astronomical units, i.e., in terms of the radius of the Earth orbit.

The duration of a parabolic transfer orbit is

$$T_{\text{p}} = 82.212 \sqrt{R_{\text{pl}} - 1} [1 + 1/3 (R_{\text{pl}} - 1)] \text{ days.}$$

At launch our target planet is located at a point in its orbit such that its time interval from spacecraft encounter is equal to the duration of the flight. The initial position of the target planet for any chosen transfer trajectory (in particular, for the Hohmann or parabolic transfer) is given by the specific Earth-Sun-planet angle. This angle regularly repeats itself with the so-called /11 synodic period. The synodic periods of Jupiter and the more remote planets are a little more than a year, since these planets move so slowly that they advance only a little in the time of one revolution of the Earth around the Sun.

TABLE 1*
HOHMANN INTERPLANETARY TRANSFER

Planet	Heliocentric approach velocity from the Earth sphere of influence, km/sec	Geocentric velocity of escape from the Earth sphere of influence, km/sec	Initial velocity, km/sec			Flight duration
			At the Earth surface	At a height of 200 km	At departure from an orbit at height 200 km	
Mercury	22,233	— 7,532	13,480	13,343	5,554	105.5 days
Venus	27,291	— 2,494	11,461	11,293	3,504	146.1 >
Mars	32,729	2,944	11,567	11,401	3,612	256.9 >
Jupiter	38,577	8,792	11,235	14,093	6,304	2 yrs 266.9 >
Saturn	40,074	10,279	13,188	15,066	7,277	6 yrs 17.6 >
Uranus	41,065	11,280	15,886	15,766	7,977	16 > 13.9 >
Neptune	41,439	11,654	16,154	16,036	8,247	30 > 224.4 >
Pluto	41,600	11,815	16,270	16,153	8,364	45 > 217.1 >

*[Translator's note: Commas in the numbers indicate decimal points.]

Effect of Ellipticity and Inclination of Planetary Orbits

We first deal with ellipticity. The eccentricity of the Earth orbit is small and on a plan view of the Solar System it is not possible to distinguish the lack of circularity of the Earth orbit. However, at perihelion of the Earth orbit, the speed of our planet is 1 km/sec greater than its speed at aphelion. Therefore, a launch at perihelion is more advantageous than a launch at aphelion. The further the target object is from the Sun, the greater is this effect. If we can reach Saturn with a certain velocity increment at Earth aphelion, we could reach Uranus with the same velocity at perihelion. Similarly, if we have a velocity increment at Earth aphelion such as to reach Uranus, at perihelion it would be sufficient to reach Pluto at its aphelion, and even much greater distances.

The ellipticity of the orbits of our target planets are less, although, naturally, it is easier to reach a planet at its perihelion than at its aphelion, other conditions being the same. This is particularly noticeable for Pluto, which is closer to the Sun at its perihelion than is Neptune. (Pluto will reach its perihelion in 1989 for the first time since its discovery in 1930; from 1979 to 1998, Pluto will be closer to the Sun than Neptune.) Since the axes of the apses of the orbits of the planets (lines joining the aphelion with the perihelion) do not coincide with the line of apses of the Earth, flights along cotangential ellipses are impossible, in principle.

However, the ellipticity of the orbits is overshadowed by the fact that all the orbits are located in different planes, although (with the exception of the Pluto orbit) they are only slightly inclined to the plane of the Earth orbit (the *plane of the Ecliptic*).*

*The large eccentricity and the large slope and large size of the Pluto orbit make this planet distinctive and give us reason to consider it to be either a form of satellite of Neptune which has broken away from that planet because of perturbations from the other planets, or as a representative of the belt of asteroids beyond Neptune which has not yet been discovered.

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The orbits drawn in plan view of the Solar System are, in fact, *projections* of the orbits on the plane of the Ecliptic. In this plane, we can construct a transfer trajectory close to Hohmann and, although it will not exactly be tangential to the Earth orbit, it will at least be half of an ellipse (the angular distance of the transfer will be 180°). However, if we break it at the point diametrically opposite the launch point (with regard to the Sun), a spacecraft moving in the plane of the Ecliptic will almost certainly not find the target plane, which may then be millions of kilometers "above" or "below" the plane of the Ecliptic.

There is an exception: when the launch point and the encounter point lie on the *line of the nodes*, i.e., the line of intersection of the planes of the orbits of the launch and target planets. But even if the Earth lies on the line of the nodes, this still does not mean that there is a launch opportunity, since the position of the target planet will very probably be unfavorable.

We present some results.

The optimal transfer trajectory is the trajectory which requires minimal velocity increment. Therefore, it can be only a little inclined to the plane of the Ecliptic. Its angular distance differs from 180° . However, a launch opportunity can be calculated approximately from the simplified model of planetary orbits; it is enough to know the required initial position of the planet (the "initial configuration angle") and to find the heliocentric longitudes of the planets from tables (they are published), at encounter. And, although we will then probably fall in the true "launch window," when it is possible to launch with the desired values of velocity increment, the true minimum (taking into account the eccentricities and the inclinations of the orbits) will occur at some other time and we may be in error by one whole month. This is the reason why, knowing that the launch of the American Jupiter 10 spacecraft occurred on March 3, 1972, by adding the average synodic period of Jupiter of 399 days (roughly 13 months), we can anticipate the next launch to Jupiter roughly in April 1973 (it took place, in fact, on April 6).

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These favorable "launch windows," which repeat roughly every synodic period, differ one from another. The best are those which fall in periods when the Earth is close to the line of nodes.

An exact calculation of the optimal launch time should account for all factors, and be performed with the help of computers.

Calculation Accounting for Earth Gravity

As yet, we have considered the motion of a spacecraft far from the Earth, outside of the so-called *sphere of influence* of the Earth. Without giving details of the theory which allows one to calculate the size of this region of space, we simply say that the radius of the sphere of influence of the Earth may be taken as roughly one million kilometers, and the boundary of this sphere can be regarded as "local infinity," i.e., infinite on the scale of Earth gravity.

We should explain that this is not a very precise formula.

All that has been said concerning free trajectories in the field of solar gravitation is valid also for motion in the sphere of influence of the Earth, under the action of Earth gravity, when all other forces can be neglected. In particular, if a spacecraft were to fly from the surface of the Earth to "infinity," i.e., the boundary of the sphere of influence of the Earth, with a residual velocity of v_{∞} , it may be moving along a hyperbola. Then the initial hyperbolic velocity v_0 is given from the formula

$$v_0^2 = v_p^2 + v_{\infty}^2 \text{ or } v_0^2 = \frac{2K_E}{r_0} + v_{\infty}^2.$$

Here, v_p is the parabolic speed (relative to Earth gravity) at distance r_0 from the center of the Earth, and K_E is the Earth gravitational parameter. For simplicity, the radius of the Earth r^* ^{/14} is taken instead of r_0 , but very often no specific distance is given. Usually, it is $r^* + 200$ km, i.e., it is assumed that the launch

vehicle finishes the burn at a height of 200 km or that the launch begins from an Earth orbit of height 200 km; this is close to the true value.

We shall now use lower case letters for distances and velocities, to distinguish between quantities in the geocentric coordinate system (relative to the Earth) and quantities in the heliocentric coordinate system (relative to the Sun).

It is clear that the velocity at infinity v_{∞} is the excess velocity which must be added vectorially to the orbital velocity of the Earth V_E , to obtain the initial velocity V_0 of heliocentric motion. In the terminology of kinematics: V_E is the transfer velocity, v_{∞} is the relative velocity, and V_0 is the absolute velocity. It is also clear that the minimal velocity increment corresponds to the minimal initial velocity, v_0 .

It is useful to compute the minimal initial geocentric speed v_0 which, after the spacecraft leaves the Earth sphere of influence, will ensure that its subsequent motion is along a heliocentric parabola, the so-called *third cosmic velocity*. As was found above, in this case, $v_{\infty} = 12.337$ km/sec. If we take $v_p = 11.186$ km/sec at the Earth surface, then $v_0 = \sqrt{11.186^2 + 12.337^2} = 16.659$ km/sec. This is the theoretical value of the third cosmic velocity at the Earth surface. Its value at a height of 200 km is more realistic:

$$v_0 = \sqrt{11.015^2 + 12.337^2} = 16.539 \text{ km/sec.}$$

The velocity v_{∞} is completely determined by the velocity v_0 acquired at launch, and does not depend on the direction. But the direction of the velocity v_{∞} on which the velocity V_0 depends appreciably and, in turn, determines the heliocentric trajectory,

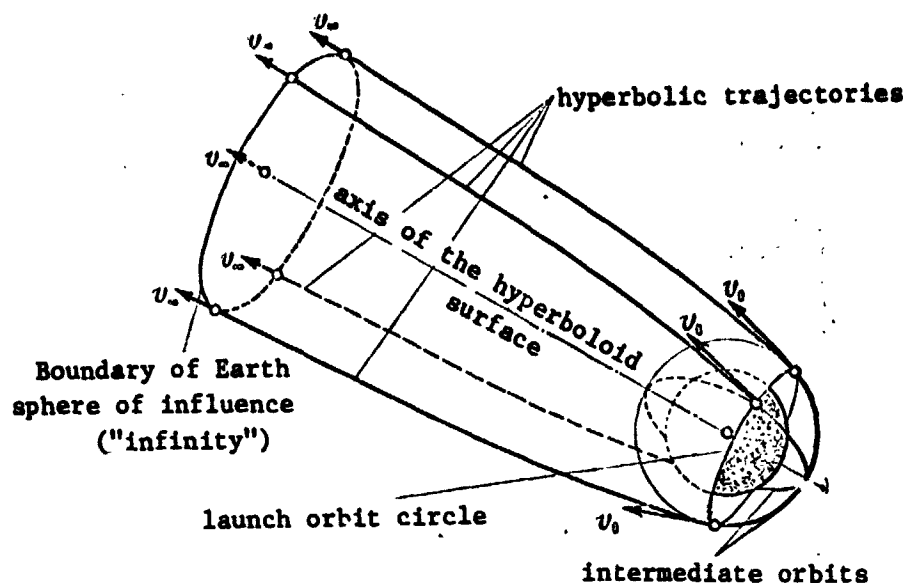


Figure 2. Hyperbolic trajectories for approach to the boundary of the sphere of influence, and circular intermediate parking orbits.

practically coincides with the asymptote* of a heliocentric hyperbola in the approach to the boundary of the Earth sphere of influence. /15
 There is an infinite number of these hyperbolas. The best of them correspond to a horizontal initial velocity v_0 . The fact is that a launch to a horizontal initial velocity v_0 requires less expenditure of rocket fuel than a steep launch to the same velocity, when the so-called *gravitational losses* are large, i.e., losses in overcoming the force of gravity.

Figure 2 shows suitable hyperbolic trajectories commencing with a burn of this kind, and forming a surface of rotation whose axis can be a trajectory (a singular straight line) for departure to the boundary of the Earth sphere of influence. Through any point of the Earth surface and this axis, one can choose a plane which intersects the surface in a suitable trajectory. From this point, one can launch a satellite into an intermediate circular parking orbit lying in this plane, the object being to transfer to a

*The asymptote of a hyperbola is the straight line to which the branch of the hyperbola approximates when one goes to infinity.

hyperbolic trajectory by increasing the orbital velocity to v_0 .

The impulse to depart from orbit must be imparted when the vehicle intersects the "orbit launch circle" in the desired direction (Figure 2). /16

Because of the Earth rotation, one can choose any intermediate satellite orbit during a period of 24 hours: to make the best use of the circular velocity of the launch site, or to go to the boundary of the Earth sphere of influence at a point close to the plane of the Ecliptic, or (which is likely to be important) to use an existing network of observational stations, and so on.

The maneuver of launch from orbit, which is quite obligatory when there is an unfavorable geographic location of the launch site, cannot be entirely free from gravitational losses, but has many fewer aerodynamic losses (losses in overcoming the resistance of the atmosphere). At least, the losses in injection into a circular orbit are unavoidable. We will evaluate these, following some foreign authors, rather arbitrarily at 1.6 km/sec, and will neglect the losses in departing from a circular orbit. In doing this, we assume that, if there is no atmospheric drag and gravitational attraction, the launch vehicle would acquire a velocity larger by 1.6 km/sec. The total, actually unattainable velocity is called the *ideal* or *characteristic* velocity.

Calculation of the Attraction of the Target Planet

Now after considering the section of flight within the Earth sphere of influence, and before considering the motion outside the sphere of influence, we will turn to the final section of the flight. The spacecraft is approaching the target planet. At a certain time in its motion, the gravity of this planet begins to be perceptible. Within a certain region surrounding the planet, called its sphere of influence, it is convenient to regard the motion, not as heliocentric (i.e., not relative to the Sun), but as planetocentric, i.e., in a system of coordinates moving along

with the center of the planet. Planetocentric motion, if one neglects perturbations, can take place only along an ellipse, a parabola, or a hyperbola.

It can be shown that the planetocentric speed of arrival of a spacecraft into the sphere of influence of any planet of the Solar System is always considerably greater than the local parabolic speed relative to the planet. Therefore, if a spacecraft does not penetrate the atmosphere of the planet, but flies by it, it moves in a hyperbolic trajectory, intersects the boundary of the sphere of influence, and passes on into a region where the influence of solar gravitation predominates. Then the emergence from the sphere of influence of the planet occurs at the same planetocentric speed as the entrance. But the directions of the vectorial planetocentric velocities at entrance and exit differ more, the closer to the planet is the flight trajectory.

Subsequent heliocentric motion can differ markedly from heliocentric motion prior to entering the sphere of influence; thus, traverse of the sphere of influence can be used for a so-called *perturbation maneuver*, which is a directional change of a heliocentric trajectory in order to reach other planets, to fly away from the plane of the Ecliptic, to fly to the Sun, and so on.

In many cases, the effectiveness of a swingby maneuver can be increased if one applies an additional velocity impulse near the swingby planet, using an onboard rocket engine (an *active swingby*).

Approximate Method of Trajectory Calculation

Even if we account for accurate data concerning the orbits of the Earth and the target plane (i.e., eccentricity and inclination), i.e., we reject the approximate model of planetary orbits and also use data on the masses of the planet, the above method of calculating interplanetary orbits is not accurate. In fact, the method entails subdivision of the whole of space into regions in which one calculates the gravitational attraction of only a single heavenly body,

the Earth, the Sun, or the target planet. The corresponding part of the trajectory is a so-called conic curve (ellipse, parabola, or hyperbola), and the method is, therefore, frequently called the *conic curve method*.

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This method can be refined in various ways. For example, without considering the boundary of the Earth sphere of influence as local infinity, we can calculate the value of the sphere radius. One can refine this radius value by taking a sphere of influence with roughly twice the radius instead (the corresponding radii in millions of kilometer are: for the Earth 0.93 and 2.50; for Jupiter 48.2 and 88.0; for Saturn 54.5 and 108.0; for Uranus 51.8 and 116.0; and for Neptune 86.8 and 194.0). In both cases, a slight improvement in initial velocity is obtained. A much more significant change is to calculate the radius of the sphere of influence of the target planet, especially for the far planets (Mercury, Venus, and Mars have comparatively small radii of the sphere of influence), when the duration of the flight is taken into account.

In actual fact, of course, a spacecraft at any portion of its flight experiences attraction from bodies on all sides: within the Earth sphere of influence (particularly close to its boundary), it experiences gravitation from the Sun, and outside the Earth sphere of influence, it experiences gravitation from the Earth (particularly close to the boundary), as well as from other planets (especially Jupiter). Strictly speaking, one should account for the gravitational attraction of all bodies of the Solar System. However, in the first place, this would lead to enormous computational difficulties, and secondly, even if it were possible, it would be useless.

We assume that we have correctly determined the value and direction of the initial velocity which must be given to the spacecraft at a particular time so that it will reach the target at the given time. However, one can communicate the required velocity only to a certain finite accuracy. There are unavoidable errors of measurement in the navigation equipment and errors arising from

malfunctions in the control devices of the spacecraft engine. Also, interplanetary distances and planetary masses are not known with sufficient accuracy. Against this background, it makes no sense to calculate the various relatively small perturbations of an interplanetary trajectory (even those caused by Jupiter).

On the other hand, an error in the initial velocity of 1 m/sec can lead to a miss of hundreds of thousands of kilometers at the target. Therefore, one must make trajectory corrections using onboard engines. A small correction impulse (at the most tens of meters per second) can be used not only to correct the position, but also to change the flight program. For example, the flight program of the Jupiter 11 spacecraft was improved when results of investigation of the radiation belt around Jupiter became available from the Jupiter 10 spacecraft. The fact is that a slight change in the direction of the velocity of approach to a planet radically alters the nature of the flight in the hyperbolic trajectory around the planet. One can pass the planet from the left or the right, or above and below it, and this not only enables one to investigate certain regions of the planetary surface, but also completely alters the fate of the spacecraft after it leaves the sphere of influence.

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Launch of an Artificial Satellite of a Planet

We have seen above that no planet is incapable of making a capture, i.e., of transferring a spacecraft flying from the Earth into an elliptical orbit appropriate to an artificial satellite of the planet, by means of its own gravitational attraction. This kind of operation requires an active maneuver: using an onboard retro engine, the speed of the spacecraft is decreased from hyperbolic to elliptic. The magnitude of the retro impulse depends on the speed of entry into the sphere of influence of the planet, the distance of the retro point from the planet center, and the eccentricity of the satellite orbit.

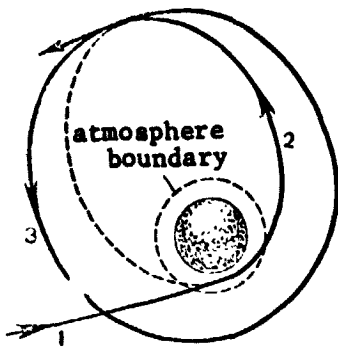


Figure 3. Use of a planetary atmosphere for injection of a satellite into orbit.

1- hyperbolic approach trajectory; 2- orbit after traversing the atmosphere; 3- orbit after imparting accelerating impulse at the apocenter of orbit 2.

If the objective is to place an artificial satellite of the planet with certainty into a circular orbit, then, for any specific planetocentric velocity at entry into the sphere of influence, one can evaluate a radius for an optimal circular orbit such that a minimum retro impulse is required (it can be shown that this impulse is equal to the local circular velocity). For example, in a transfer to Jupiter along a Hohmann trajectory, the radius of the optimal orbit is equal to 115

Jupiter mean radii, or 8 million

kilometers. Such a transfer requires a retro impulse of 4 km/sec, while 18 km/sec is required to transfer to an orbit near the upper edge of the atmosphere, under the same conditions. The gain is very great, but there is little merit in placing a satellite of Jupiter at a distance of more than 20 times the distance of the Moon to the Earth. In practice, therefore, one uses greatly elongated elliptical orbits. /20

A transfer to an elliptical orbit can also be accomplished using the planet atmosphere (aerodynamic deceleration). The aerodynamic drag replaces the retro impulse of the rocket. However, the pericenter of the orbit (the closest point to the planet) is now within the atmosphere (Figure 3). In order to withdraw it from the atmosphere, one must add a burn impulse at the most remote point, the apocenter.

There is also a somewhat different way to use aerodynamic braking. A spacecraft that has aerodynamic lift can skip up in the atmosphere and obtain a horizontal accelerating impulse at the maximum height of the skip, adding to its velocity and reaching a

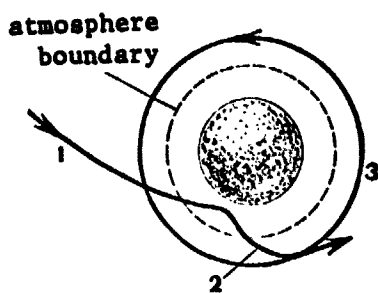


Figure 4. Use of skip-up from the atmosphere to inject a satellite into orbit.

1- hyperbolic approach trajectory; 2- skip-up trajectory; 3- satellite orbit after receiving accelerating impulse at the maximum skip-up height.

local circular velocity (Figure 4). /21
The skip-up maneuver involves a minimum impulse.

Deceleration using aerodynamic lift can also be used to perform a lateral maneuver to inject a satellite into an orbit in a plane differing from the approach trajectory.

Unfortunately, part of the energy saved by using the atmosphere is lost because of the increased expenditure of fuel in preceding stages of the flight, since a spacecraft entering the atmosphere must be provided with a heat shield, i.e., it must have an increased mass. Finally, these maneuvers require the spacecraft to accurately enter a narrow atmospheric corridor, which is not easy to accomplish.

These disadvantages are avoided in a compromise method of satellite launch in which a small rocket impulse is given in rarefied layers of the atmosphere to place the satellite into an elliptic orbit with a high apocenter. In passing repeatedly through the upper layers of the atmosphere, the satellite transfers to an orbit with a lower apocenter, decreasing at each revolution. When the apocenter becomes low enough, a small engine impulse at the apocenter will raise the pericenter. Thus, the spacecraft can be injected into a low and almost circular orbit. According to calculations of the Soviet scientist, N. A. Eismont, who has proposed this method for launching a satellite of a planet (*Journal of Space Research*, Vol. 10, No. 2, 1972), the gain in velocity in launching a satellite of Jupiter can be as large as 12.7 km/sec, if the first elliptic orbit has an eccentricity of 0.8.

A Brief Review of Rocket Dynamics

In this section we shall consider thermochemical and nuclear heat engines, as far as they pertain to rocket engines.

The duration of each active section of interplanetary flight (sections in which the engines operate) is measured in minutes, and even in seconds (for trajectory corrections). It is, therefore, natural to regard any change in velocity as a short duration impulse (or discontinuity), bearing in mind the very large duration of the flight. It is important to stress that, in contrast to the velocity of the spacecraft, the burn can be calculated in different reference systems, and that the velocity impulse communicated by a rocket motor is an absolute quantity, independent of the coordinate system. /22

The complexity of a space operation is determined to a considerable degree by the number of impulses. An ordinary flight, including a descent to a planet, is a one-impulse operation: a descent with retro engine deceleration to a landing is a two-impulse operation; launch of a satellite of a planet (with retro engine deceleration) is also a two-impulse operation; return to Earth from an orbit around the planet is a three-impulse operation; and return to an orbit around the Earth is a four-impulse operation. The number of pulses here does not include correction impulses. The launch from the Earth is considered as one impulse.

The sum of the values of all the impulses during the time of the entire operation is called the total characteristic velocity. The total characteristic velocity also includes gravitational and aerodynamic velocities during launch from the Earth, which will be assumed to be 1.6 km/sec in what follows. We will neglect gravitational losses during the launch from any orbit, with rocket engine injection into orbit and other maneuvers, since, as a rule, the impulses are communicated in directions which are roughly perpendicular to the forces of gravitation, or are very remote from the bodies generating these gravitational forces.

We assume that a rocket system consists of n stages and that the structural features of each stage are such that each has a constant ratio s of the initial mass of the stage, including its fuel, to the mass without fuel. We shall also assume that the discharge velocity of the combustion products is the same for all stages, and is w . If we denote the payload of the system by m_p and its initial mass by M_0 , then the relative payload $P = M_0/m_p$ can be expressed as

$$P = e^{V/w} \left(\frac{s-1}{s - e^{V/nw}} \right)^n$$

where V is the total characteristic velocity. The value of s can be quite large (of the order of 15 or 20). The number of stages n need not be equal to the number of impulses, by any means. The discharge speed w can, at best, reach a value of 5 km/sec for liquid rocket engines using hydrogen as fuel and oxygen, or better fluorine, as oxidizer. For nuclear rocket engines with a solid motor, one can attain $w = 10$ km/sec, with a liquid motor — 20 km/sec, and with a gas motor — 30 to 70 km/sec. These numbers are essentially hypothetical, being based on foreign publications. /23

Knowing the total characteristic velocity V , we can assign the values s , w , and n , and calculate P from the formula given. If we now assign the payload m_p , we can find the initial mass $M_0 = m_p P$ and can assess the difficulties of constructing any spacecraft to be launched from orbit.

When the mass M_0 cannot be injected into orbit by existing (or planned launch vehicles or by multi-journey space transport vessels (space shuttle), one must plan to assemble a spacecraft with a rocket system in orbit. Naturally, the assembly orbit should coincide with an intermediate parking orbit around the Earth.

It is not expedient to calculate the mass of the launch vehicle launched from Earth by the above formula, since the stages of this rocket cannot have the same values, either of w or of s , for a number of reasons.

We should stress that the payload includes not only scientific apparatus and all the supply system (e.g., the telemetry system), but also the guidance and navigation systems.

Flights with Low Thrust

For flights to remote planets, a particularly efficient form for future propulsion is electric rocket engines which impart accelerations of the order of 10^{-5} to 10^{-3} g to a spacecraft ($g = 9.8 \text{ m/sec}^2$ is the acceleration due to Earth gravity). The only energy source for such engines when they are remote from the Sun is a nuclear reactor, although solar cells can be used in the initial stages.

The extremely low thrust of electric rocket engines dictates that they be used only after injection into orbit around the Earth. When acting continuously (or continuously at intervals), these low thrust engines will first remove the spacecraft from the sphere of influence of the Earth in a spiral trajectory, and will then accelerate it along a heliocentric spiral. If the problem is not to fly by the planet but to inject into an orbit around a target planet, then deceleration begins at the end of the heliocentric section, to allow capture of the spacecraft by the planet (such a capture is impossible in the case of a flyby). Then the spacecraft descends in a planetocentric spiral (the small thrust is directed opposite to the velocity) to a low orbit.

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Interplanetary flights with low thrust allow a considerable increase in payloads. If we are considering injection into an orbit around a planet, the flight duration can be considerably shortened (not only is the time to descend along a spiral reduced, but also the need to decelerate before descending to the planet is eliminated).

An additional reduction of the time can also be obtained by avoiding an ascent along a geocentric spiral and using only the ordinary impulsive departure from an orbit around the Earth to local infinity. Although some part of the advantage in payload is then lost, the control of the flight is greatly simplified.

CHAPTER 2

DIRECT FLIGHT TO THE FAR PLANETS

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Flight to Jupiter

We first consider flight to the planets without flyby of another planet, i.e., so-called *direct flight*.

The initial velocity relative to the Earth surface to reach Jupiter in a Hohmann trajectory is 14.238 km/sec. The flight takes two sidereal years and 276 days (997 days).

An increase in the velocity of departure from Earth reduces the flight time. But even at the third cosmic velocity of 16.54 km/sec, the flight takes one year and 39 days.

A synodic period of revolution of Jupiter is 399 days, i.e., the season favorable for flight to Jupiter begins each year with a delay of roughly one month (e.g., March 1972, April 1973, May 1974, June 1975, August 1976, September 1977, etc.). The most favorable seasons are those with departure at the beginning of January and the beginning of June, when the Earth is close to the line of nodes of the orbit of Jupiter. The January opportunities are particularly favorable, since the Earth in January is close to its perihelion, where its velocity is 1 km/sec greater than at aphelion, which occurs in June. Launches at the January opportunities

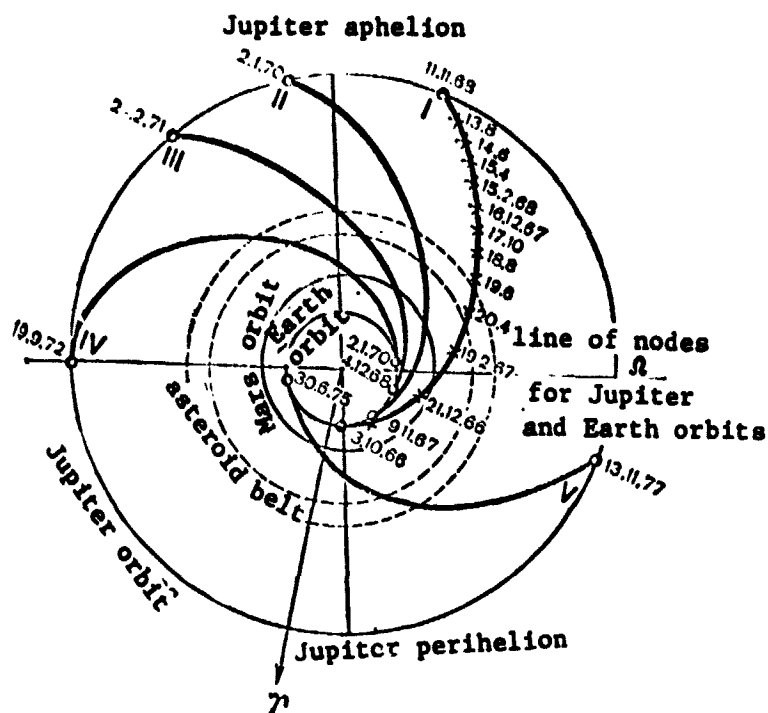


Figure 5. Projections on the plane of the Ecliptic of the optimal trajectories of a flight to Jupiter. (R. K. Kazakova, V. G. Kiselev, and A. K. Platonov, Kosmicheskiye Issledovaniya, Vol. 6, No. 1, 1968).

have an angular range close to 180° , the greatest flight time, the least inclination of the plane of the flight trajectory to that of the Ecliptic, and the least initial velocity.

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Figure 5 shows five flight trajectories to Jupiter which require a minimum initial velocity for each favorable opportunity. Of the trajectories shown, the least initial velocity is required for trajectory II (launch on November 9, 1967), since the launch occurs far from the line of nodes, when Jupiter is close to its aphelion. Trajectory IV is closest to a Hohmann transfer (launch on January 2, 1970). For trajectory II, the flight time is 791 days; the angular range is 162° ; the velocity at exit from the Earth sphere of influence is 9.526 km/sec; the inclination to the Ecliptic is 4.14° . The corresponding data for trajectory IV are: 989 days, 179.3° ; 8.673 km/sec; and 0.33° .

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On its path to Jupiter, a spacecraft moving with minimum velocity intersects the orbit of Mars in three months, and traverses the astroid belt in the period five to eight months from launch.

At the beginning of the geocentric part of the flight, a velocity deviation of 1 m/sec causes a deviation near the planet, in the "map" plane (perpendicular to the line of sight) of roughly 120,000 km for most of the trajectories of Figure 5, and a deviation of 340,000 km for trajectory IV. In the middle part of the flight, a deviation of 100,000 km can be compensated for by a correction impulse of the order of 3 m/sec. To change the flight time by 12 hours, a correction impulse of about 10 m/sec is needed. (Such an impulse can be applied in order to ensure good radio communication conditions at the time of approach to Jupiter: the time for two-way communication from Earth to Jupiter is 12 hours.)

Entry into the Atmosphere of Jupiter

The atmospheric entry speed without allowing for rotation is: for a Hohmann transfer 60.7 km/sec, and for a parabolic transfer — 62.9 km/sec. If the atmospheric entry occurs near the Jovian equatorial plane and is directed with the rotation around the axis, the relative entry speed decreases by the amount of the circular speed of a point on the Equator (12.6 km/sec), and becomes roughly 50 km/sec. This speed is very great and presents a problem.

The maximum loading in vertical atmospheric entry, according to calculations of the American author Whingrow, is 3700 g (for Venus, it is 300; and for Mars — 50). For the loading not to exceed 100 g, the entry angle into the Jovian atmosphere must not exceed 1 or 2°. The entry corridor with no lift must have zero width if the loading limit is taken as 10 g. In other words, it is impossible to perform an entry in a corridor of this kind. With an aerodynamic lift coefficient of 1, the entry corridor width is 83 km.

A spacecraft to survive deep penetration into the Jovian atmosphere must be designed for tremendous pressures, and probably for very severe chemical and electrical effects. A simpler problem, according to an American paper, is to penetrate to the lower level of the clouds. Here a probe must pass through a layer of clouds composed of ammonia crystals, ice crystals, water drops, and finally drops of ammonium chloride, to reach a depth of roughly 150 km (for the "nominal" atmosphere model, i.e., not the "cool," nor the "warm" model atmosphere) where the pressure is postulated to be 17 bar, and the temperature 425° K (the values differ for the other atmospheric models). An entry probe is separated at Mach number of 0.7 (ratio of speed to the speed of sound). This occurs after 20 seconds (for almost direct entry) or after 60 seconds (at entry angle of 15°) after reaching a loading of 0.1 g. The subsequent descent occurs by parachute, which separates after the probe descends through the water drop clouds, so that the probe may reach the ammonium chloride clouds.

An Artificial Satellite of Jupiter

The retro impulse which must be imparted to a spacecraft to inject it into a low satellite orbit around Jupiter (we have in fact an orbit passing through the extreme edge of the atmosphere) is 18 km/sec for a Hohmann transfer. The total characteristic velocity at launch from a low orbit around the Earth (height 200 km) is 24 km/sec. As a calculating using the formula presented above shows, with a discharge speed of $w = 4$ km/sec and with $s = 15$, the payload is $P = 2511$ for a three-stage launch vehicle, and $P = 1266$ for a four-stage vehicle. Even with a payload $m_p = 0.2$ tons, the initial mass of a four-stage vehicle must be more than 250 tons, i.e., it would take at least two Saturn V launch vehicles, or not less than 10 flights of the space shuttle to put this weight into an orbit around the Earth.

At a future time, when all the data on the Jovian atmosphere becomes known and a very precise entry into Jupiter will be permissible, it will be possible to use aerodynamic deceleration in the atmosphere. After entry into the atmosphere, as has been mentioned, a further rocket impulse is added, and the total characteristic velocity for the entire mission will probably exceed the third cosmic velocity. The possibility of a gradual descent has been mentioned already, using the method of "braking ellipses" to descend to a low satellite orbit of Jupiter. /29

The optimum circular single-impulse orbit for an artificial satellite of Jupiter, using a theoretical Hohmann transfer, has a radius of 115 mean Jupiter radii (8 millions of kilometers). Its period of a revolution is 145 days. The retro impulse, equal to the local circular velocity, is 4 km/sec. It is possible to launch mass of 0.5 tons into such an orbit, if a two-stage vehicle of mass 9 tons is launched from a low Earth orbit. It can be injected into orbit using a single journey of the space shuttle. However, this particular orbit is still very remote from Jupiter, the actual planet. It is clearly advantageous to launch satellites of Jupiter into elliptical orbits with pericenter quite close to the planet. For example, if the hyperbolic velocity near the Jovian surface were reduced by means of a retro impulse to 55 km/sec, then the apocenter of the satellite orbit would be located at a height of 4.3 Jupiter radii above the surface of the planet (roughly 300,000 km). The period of revolution in this orbit is 1665 hours. With the previous assumptions, a vehicle of mass 18 tons must be launched from Earth orbit. This is roughly the injection capability of the Soviet Proton launch vehicle.

If the flight to Jupiter is along a heliocentric parabola, the retro impulse for transfer to a low circular orbit is 20 km/sec. The optimum circular satellite orbit then has a radius of only 12 mean Jupiter radii (830,000 km); the retro impulse (equal to the local circular velocity) is considerable, 12 km/sec.

The idea has been put forward to use the gravitational attraction of a large Jovian satellite (e.g., Ganymede) for a perturbation retro maneuver, in order to reduce the rocket impulse for injection into orbit around Jupiter. /30

A stationary satellite orbit of Jupiter must have a radius equal to 2.3 mean Jupiter radii (162,000 km). In theory, a satellite in this orbit would hold its position above a specific point of the planet surface (more correctly, above the atmospheric layers whose rotation we observe). Such a satellite would always see 56% of the surface of one hemisphere of Jupiter.

The orbits of artificial satellites of Jupiter will experience a severe perturbing effect from the nonspherical nature of its gravity. The oblateness of Jupiter is very great (it can be seen by eye in a telescope): its polar radius is 4.5 thousand kilometers less than its equatorial radius. The gravitational attraction of the large natural satellites of Jupiter will also be important.

Flight to the Natural Satellites of Jupiter

The natural satellites of Jupiter are heavenly bodies of great interest. Four of them, Io, Europa, Ganymede, and Callisto, are huge. The largest, Ganymede, has a radius of 2500 km; the escape velocity (parabolic velocity) at its surface is 2.9 km/sec. Flights to these satellites will not be like simple maneuvers, as would be the case of flight to the tiny satellites of Mars — Phobos and Deimos. Probably a rocket deceleration will be provided, although an atmosphere is thought to exist for Ganymede and Callisto, and an atmosphere was apparently observed for Io during the flight of the American spacecraft Pioneer 10.

In this case, one does not need to have one impulse to equate the speed of the spacecraft to that of the satellite (clearly, the impulse would be applied at the boundary of the sphere of influence of the satellite), and another impulse to slow the velocity of

descent of the satellite. It is more convenient on an energy basis to replace these two operations by one, in fact, and to arrive with the theoretical speed of approach from the Moon or the planets. If we consider that a satellite of Jupiter must be located in an orbit, the directions of the planetocentric velocity v_{sp} of the spacecraft (at the boundary of the satellite sphere of influence) and of the velocity v_{sa} of the satellite coincide, and if, in addition, we neglect gravitational losses, then the required retro impulse v_{ret} is given by

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$$v_{ret} = \sqrt{(v_{sp} - v_{sa})^2 + v_p^{*2}},$$

where v_p^* is the parabolic velocity at the surface of the satellite.

However, before we go on with a landing on an artificial satellite, we must first inject the spacecraft into a parking orbit located in the equatorial plane of Jupiter (this plane contains the orbit of the five satellites closest to Jupiter, including the four giant satellites). Only when a favorable time comes is it possible to make a Hohmann transfer from this orbit to the orbit of the Jovian satellite and the next maneuver which requires a velocity impulse as given above.

Flights to Saturn, Uranus, Neptune, and Pluto

When considering direct flight to Saturn, Uranus, Neptune, and Pluto, we expect to have large rocket burns. The point is that the minimum speed for reaching these planets are comparatively large: naturally, they do not exceed the third cosmic velocity. An unattractive feature is that the duration of the flights is very great.

For example, the minimum velocity to reach Saturn (in the simplified model of planetary orbits) is 1 km/sec greater than the corresponding velocity for Jupiter, but the flight duration is six years. Increasing the departure velocity up to the third cosmic value reduces this time to somewhat more than two years.

The minimum velocities for flight to Uranus, Neptune, and Pluto are very little different, since they are already close to the third cosmic speed. But the duration of these flights are enormous. A flight to Pluto (at its mean distance) on a parabolic trajectory, takes more than 19 years.

In 1989, Pluto will be at perihelion, at a distance of 4.4 billion kilometers from the Sun, in fact, closer to Neptune. A direct flight to it, timed for this date, with launch at the third cosmic velocity (more correctly, somewhat less because of the inclination of Pluto orbit), would take 13 years, i.e., it would begin in 1976. We shall see below how we can reach Pluto earlier, by departing later, and with less initial velocity.

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The synodic periods of these planets are somewhat greater than one year. An opportunity favorable for a flight to Saturn occurs annually, with a delay of two months. For Uranus, Neptune, and Pluto, the delay ranges from 4 to 1 days. Spacecraft for flight to the Jupiter group of planets can be launched each year, and then one must await the results of the experiment through long weary years.

As was true for flights to Jupiter, the motion within the spheres of influence of Saturn, Uranus, Neptune was determined mainly by the gravitational attraction of the respective planets, and only to a slight extent by the entry velocities into the planets spheres of influence.

The entry speed into the Saturn atmosphere, following a flight on a Hohmann trajectory, is 36.6 km/sec, and for a flight on a parabolic trajectory, it is 38.8 km/sec. An atmospheric entry in the equatorial plane in an easterly direction would reduce these values by more than 25%, since the circular speed here is 10 km/sec, but the Saturn rings constitute a problem for this kind of entry.

The entry speeds into the atmospheres of Uranus and Neptune are, respectively, 22.0 (23.9) and 24.0 (25.2) km/sec (the figure outside the bracket is for Hohmann transfer, and within the bracket for parabolic transfer). The circular velocity of points on the equator at Uranus and Neptune are, respectively, 3.9 and 2.5 km/sec, and one can also make use of these.

Similarly, the impulses for transfer into the orbits of low artificial satellites also depend very little on the transfer trajectories (the figure outside the bracket is for Hohmann transfer, and inside the bracket — for parabolic transfer): for a satellite of Saturn 11.0 (13.2), of Uranus 6.8 (8.7), and for Neptune 7.3 (8.4) km/sec.

The Saturn rings evidently prohibit the launch of artificial satellites into orbits passing at distances of between 0.5 and 1.25 of the mean Saturn radius from the surface of the planet. The prohibited orbits have rotational periods of from 4 to 14 hours (these are the periods of revolution around Saturn of the particles composing the rings). In particular, a stationary orbit is prohibited (the period of revolution of Saturn about its axis is 10 hours, 14.5 minutes). /33

Stationary satellites of Uranus and Neptune, whose orbits have radii of 2.6 and 3.4, respectively, of the mean planet radius, could provide constant observation of 61 and 71%, respectively, of one hemisphere of these planets.

Saturn, Uranus, and Neptune possess large satellites which are themselves objects for independent investigation. Of particular interest is the Saturn moon Titan, which has a dense atmosphere that could, evidently, be used for deceleration in a landing.

The lack of definition in the mass and radius of Pluto makes it impossible to present reliable theoretical data on the orbits of its artificial satellites.

Low Thrust Flight to the Giant Planets

Judging from the published work, low thrust flights to the distant planets have emerged from the theoretical stage and are now in the stage of preliminary development. Both flights with solar-electric and with nuclear-electric propulsion are being considered. These plans involve departure from Earth orbit using liquid rocket motors, and not low thrust motors. Here one obtains a certain "residual hyperbolic velocity at infinity" which, although insufficient to reach the target planet, involves motion along an abbreviated spiral around the Earth.

We shall present results published in two American papers in 1972.

Departure from the sphere of influence of the Earth can be accomplished using the "Centaur" launch vehicle (mass of 17.2 tons) injected along with a spacecraft (total mass, 25.5 tons) into an Earth orbit of height 500 km by the space shuttle. The geocentric injection velocity is only 2.9 km/sec, all told. A universal spacecraft is used, a cylinder of length 17 m and diameter 1.4 m. The electric propulsion motor is located in the middle of the spacecraft and creates a transverse thrust. The power of the nuclear thermionic energy source when the engine is switched on is 120 kW, and the specific impulse of the electric rocket engine is 5000 seconds. The instrument section of mass 700 kg is located in one of the ends /34 of the cylinder. The entire flight, beginning in 1986, lasts 900 days: 240 days of acceleration, 320 days of coasting flight, and 340 days of deceleration prior to entry into the sphere of influence of Jupiter and descent in a spiral path (it continues for 158 days) into an orbit of radius $5.9 r^*$ (r^* is the mean Jupiter radius), which corresponds to the radius of the orbit of the moon Io. Altogether, in 18,000 hours of operation, the electric propulsion engine uses 4.2 tons of mercury.

In cases where launch along a low thrust spiral is not planned, in comparable conditions, one can only achieve injection into elongated elliptical orbits with large enough ($6 r^*$) pericentral distance, since the liquid rocket engine can provide only a small retro impulse.

For example, a satellite of mass 762 kg (including 162 kg of structure) can be injected into an orbit around Jupiter with a distance at pericenter of $6 r^*$ and $37 r^*$ at apocenter (period of revolution is 9.95 days), with a flight duration of 840 days, if the launch is accomplished using the five-stage "Titan-Centaur" launch vehicle, capable of launching a spacecraft of mass 2195 kg out of the Earth sphere of influence with geocentric velocity of $v_\infty = 7$ km/sec. The power of the electric propulsion unit is 15 kW, and the specific impulse is 3000 seconds. The thrust can be deflected from the Earth-spacecraft line to an angle of $65 - 68^\circ$. The electric rocket unit and the solar cells are separated after 300 to 400 days of operation. The retro impulse of 2.491 km/sec for transfer to a satellite orbit is imparted by an onboard liquid rocket engine with specific impulse of 372 seconds.

As regards the other planets, we present the results of a calculated mission to inject the above universal spacecraft into an orbit around Uranus with a payload of 700 kg: $v_\infty = 2.8$ km/sec, flight duration — 1950 days (including 16 days of spiral descent to an orbit of radius $16 r^*$, r^* being the radius of Uranus), and the electric rocket motor uses 4.3 tons of mercury in 21,000 hours of operation.

Flight of the same spacecraft towards Neptune, without injection into a satellite orbit, requires a total of only 1650 days (4.5 years), with $v_\infty = 3.6$ km/sec and a mercury expenditure of 3.6 tons in the electric propulsion unit after 15,000 hours of operation.

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It is interesting to compare this with a flight using a liquid engine. For launch from a low orbit with minimum of 8.2 km/sec a flight to Neptune requires 31 years (see Table 1). Assuming an exhaust velocity of $w = 4$ km/sec (oxygen-hydrogen fuel), we find that the relative initial mass P of a single-stage rocket is roughly 14. With an assumed payload of 700 kg, the initial mass departing from orbit must be about 10 tons. This is less by a factor of 2.5 than the mass which can be injected into orbit by the space shuttle, but the loss of time is enormous. A flight with the third cosmic velocity (departure from orbit with a speed of 8.8 km/sec) would take up to 13 years.

CHAPTER 3

PERTURBATION MANEUVERS

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Via Jupiter to the Sun

It is well known that the mass of Jupiter is considerably more than the total mass of the remaining planets of the Solar System. It is, therefore, not surprising that the powerful gravitational attraction of Jupiter can be used for various maneuvers in interplanetary flight. In fact, in a flight through Jupiter sphere of influence, the heliocentric motion of a spacecraft experiences a "gravitational shock" equivalent to a velocity impulse which can reach a maximum value of 42.5 km/sec. Naturally, this impulse cannot be used freely, in contrast with a rocket impulse, since its magnitude and direction are determined by the velocity and by the line of approach into the sphere of influence of Jupiter, but such a large value of maximum velocity increment gives rise to great possibilities.

We shall see below that Jupiter is capable of working wonders in its role as a "trajectory generator." It is true that, to accomplish these wonders, one must first fly to Jupiter, but it is our good fortune that the largest of the giant planets is located closest to us.

Saturn has also a great deal to give, but it is considerably further away and has a mass half as great as Jupiter. However, the assistance that Saturn can add to that of Jupiter is by no means without value. We shall address this question later, but now we turn to Jupiter alone.

It is well known that a direct flight to the Sun is a considerably more difficult objective for space rocket engineering, from the viewpoint of the energy expenditure, than to reach the most distant region of the Solar System. The reason is that the orbital motion of the Earth, which is the starting point of any flight to external planets, must first be overcome in a flight to the center of the Solar System. For a spacecraft to begin to fall towards the Sun, its velocity at exit from the Earth sphere of influence must be equal to the velocity of the Earth (29.785 km/sec), but in the opposite direction. The initial velocity increment relative to the Earth surface for this must be $\sqrt{29.785^2 + 11.862^2} = 31.816$ km/sec. This quantity is sometimes called the *fourth cosmic velocity*. /37

A value only a little less, i.e., 29.151 km/sec, is the necessary velocity of departure from the Earth surface in order to reach the rear edge of the Sun along a semi-ellipse. The duration of the flights to the Sun is practically the same in both cases: 64.57 and 65.05 days.

In truth, there is a method of reaching the Sun requiring expenditure of energy only a little more than what is necessary to leave the Solar System. This is the so-called "transfer via infinity." A spacecraft is sent on an ellipse tangent to the Earth orbit, to the distant regions of the Solar System, where its velocity becomes so small that it can be completely cancelled by a small rocket impulse at aphelion, whereupon it begins to fall towards the Sun. However, the duration of the entire episode is some tens of years. For example, at aphelion at 20 a.u.* (behind the orbit of Uranus), the approach to the Sun takes 33 years from launch (including 16 years of straight-line fall).

*1 a.u. is the astronomical unit, i.e., the average distance of the Earth from the Sun, equal to 149,600,000 km.

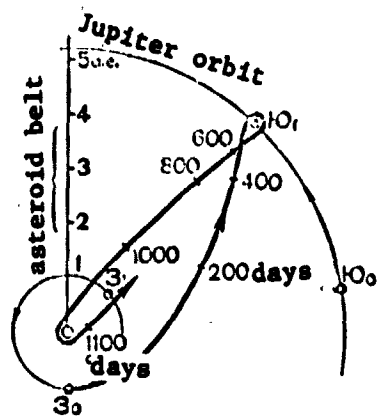


Figure 6. Flight to the Sun via Jupiter.

It turns out that, with the help of Jupiter, we can reach the near vicinity of the Sun with low expenditure of energy and in an acceptable time. In this case, two of the largest bodies of the Solar System become the objects of investigation in a single flight.

Figure 6 shows one of the many 38 possible published trajectories for Jupiter swingby, requiring an initial characteristic velocity of

16.5 km/sec (including losses of 1.22 km/sec). Passing at a distance of 5.3 Jupiter radii from the center of the planet, a spacecraft leaves the Jovian sphere of influence, describes a loop around it strongly reminiscent of the loop in a flight around the Moon, and is then thrown off towards the center of the Solar System. Three years after launch, it passes at a distance of 0.2 a.u. from the Sun with velocity 298 km/sec. An attempt to reach these distances directly would require an initial velocity at the Earth surface of 16.841 km/sec (without allowing for losses).

It has been shown that, with an initial characteristic velocity of 16.8 km/sec (taking into account losses assumed to be 1.22 km/sec), a Jupiter swingby at the necessary distance from the surface will achieve a flight to the Sun. On the other hand, an approach to the Sun at distance 0.1 a.u., requires an initial velocity of 20.421 km/sec.

These questions can be posed: can one bring back to the Earth a spacecraft which has performed a Jupiter swingby, as if it was being intercepted during its mission to the center of the Solar System? But it has been shown that it is quite impossible for the Earth to be at all close to the point of interception at the time the spacecraft intercepts the Earth orbit. For example, Figure 6 shows an intersection of Earth orbit occurring three years after

departure from Earth, and therefore, at the time the Earth is located near the point E_0 , and, therefore, it should be to the left of E_1 (E_1 is the position of the Earth at the time of the Jupiter swingby).

If the entire flight were considerably longer, as for example, in flights around the more distant planets, it would clearly be easier to arrange for the flight duration to coincide with the motion of the Earth and to ensure return of the probe to Earth.

We note that a flight near the Sun can also be achieved by a swingby around Saturn or the planets beyond it, but these missions are unattractive because their duration is so great.

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Flight Out of the Plane of the Ecliptic Via Jupiter

There is great scientific interest in investigating the properties of interplanetary space far from the plane of the Ecliptic. It can be considered that this is the plane containing the main mass of cosmic dust. A flight far from the plane of the Ecliptic would possibly give a great deal of data to assess the origin and evolution of the Solar System.

However, deviation out of the plane of the Ecliptic requires considerable expenditure of energy. The expenditure differs a great deal, depending on the region outside the plane of the Ecliptic that we wish to investigate.

It is very easy to penetrate into regions remote from the plane of the Ecliptic by accomplishing this at the edge of the Solar System. To do this, it is enough to place an artificial planet in an outside elliptical orbit, inclined at some angle to the plane of the Ecliptic. Even a slight inclination will place a spacecraft tens of millions of kilometers from the plane of the Ecliptic, when it is a great distance from the Sun.

But it is much more difficult to penetrate the space "above" and "below" the Sun. Let us suppose that we aim to place a spacecraft in a circular orbit perpendicular to the plane of the Ecliptic. Moving in such an orbit, a spacecraft must meet the Earth half a year following launch. Calculation shows that the initial velocity must be 43.58 km/sec, i.e., it is considerably greater than the fourth cosmic velocity.

A flight in an elliptic orbit lying in the plane perpendicular to the Ecliptic, but with its perihelion located beyond the Sun and close to its surface, would require an initial velocity somewhat greater than the fourth cosmic velocity, but the maximum distance of the spacecraft from the plane of the Ecliptic (midway from the Earth to the Sun) would be 0.068 a.u., i.e., 10 million kilometers. This is a very small value on the scale of the Solar System, and yet the launch velocity is almost unattainable. /40

But it is rather simple to investigate regions lying at many millions of kilometers "above" and "below" the Earth orbit. To place an artificial planet in a circular orbit of radius 1 a.u., with its plane inclined at angle i to the plane of the Ecliptic, we require a geocentric velocity of departure equal to $v_0 = 2V_E \sin(i/2)$. (V_E is the orbital velocity of the Earth.) For $i = 10^\circ$, we find that $V_E = 5.19$ km/sec, whence $v_0 = 11.19 \sin 5.19^\circ = 12.2$ km/sec. We can see that the velocity of departure from Earth is small, but it allows a spacecraft to reach a maximum distance of 26 million kilometers from the Earth in three months from launch. We note that such an orbit, lying close to the Earth orbit (although beyond the sphere of influence of the Earth), must be subject to a considerable perturbation from our planet.

An additional deviation from the plane of the Ecliptic can be achieved in this case if a solar-electric rocket engine is carried on board. Investigations along these lines were published in the USA in 1971.

It seems that it would be easy to accomplish a departure from the plane of the Ecliptic by a "transfer through infinity." A small velocity at a distant aphelion can easily turn a spacecraft through 90° , by means of a small rocket impulse. For example, for an aphelion of distance 40 a.u. from the Sun (the mean distance of Pluto), it is enough to impart a velocity of 1.4 km/sec to a spacecraft so that its Hohmann orbit is turned, without alteration, through 90° around the Sun-aphelion line. Here the maximum distance from the plane of the Ecliptic is 6.32 a.u. = 945×10^6 km and will be located roughly above the orbit of Uranus. The total characteristic velocity is 17.7 km/sec (relative to the Earth surface, losses not being taken into account). However, it takes 45 years of flight to aphelion and half as long again to reach the maximum distance from the plane of the Ecliptic.

Now we consider what assistance Jupiter can give.

In a survey paper at the Symposium of the American Astronomical Society in 1965, St. Ross showed that, for motion in a trajectory close to Hohmann, and with the required entry into the sphere of influence of Jupiter, the motion following emergence from the sphere of influence can be deviated from the plane of the Ecliptic by an angle somewhat more than 23° . However, one can achieve a rotation through an angle of 90° , but to do this requires a large velocity of departure from Earth. /41

The problem of emergence from the sphere of influence of Jupiter in a plane perpendicular to the Ecliptic, and the subsequent flight at a given distance from the Sun has been examined in detail by the Soviet scientists N. G. Khavenson and P. E. El'yasberg, in a paper published in 1972. Here the closer the approach to the Sun, the less is the departure from the plane of the Ecliptic, but the smaller is the necessary departure velocity. For example, in the 1975 opportunity for a given perihelion distance of 0.05 a.u., the maximum departure from the Ecliptic at a northerly direction is 0.45 a.u., and requires a geocentric velocity of departure from the

sphere of influence of the Earth of 11.06 km/sec; for departure in the southerly direction, the values are 0.54 a.u. and 11.09 km/sec. The corresponding data for a perihelion distance of 0.2 a.u. are 0.95 a.u. and 11.16 km/sec; 1.03 a.u. and 11.22 km/sec. These velocities correspond roughly to the minimum velocity for reaching Uranus. The maneuver is accomplished by a swingby of Jupiter at distances of 460 to 510 thousands of kilometers from the center of the planet. The approach to the Sun occurs 3.2 to 3.3 years after launch. The optimum launch date in 1975 is June 27th.

In the above report, St. Ross stated that, instead of rotating the plane of flight to 90° , and then flying "above" or "below" the Sun, it is better to attempt to maximize the heliocentric velocity component of emergence from the sphere of influence of Jupiter perpendicular to the plane of the Ecliptic. For the same Earth departure velocity, this would allow departure from the plane of the Ecliptic by 6 a.u. greater than for the 90° rotation.

Via Jupiter or Saturn to the Haley Comet

The majority of comets have very elongated orbits, often steeply inclined to the plane of the Earth orbit. Some of the comets pass around the Sun in a direction opposite to that in which the planets rotate around the Sun (in this case, the angle of inclination is greater than 90°).

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Therefore, even a simple flight past a comet is no simple task in space engineering. In fact, to avoid excessive expenditure of energy, it is desirable that the transfer trajectory lie in the plane of the Ecliptic, for which the necessary velocity is directed to the point of intersection of the orbit of the comet with the plane of the Ecliptic. But arrival at this point at the moment when the comet core passes through it is impossible because of the lack of "congruence" of the motions of the Earth and the comet. However, there are many comets, and so there is a certain choice of objects to be investigated.

The truth is that a flight past a comet can be of no value if the flyby speed is too large, since then the remote cameras would simply fail to record anything. This would be the case with a countermotion of the comet. For example, at the beginning of 1986, a spacecraft could fly by Haley's comet with a velocity of 70 km/sec, while for certain work, the requirement as regards limiting relative velocity is 16 km/sec, if the distance between the comet core and the spacecraft is less than 5000 km. An event which is much more interesting in a scientific sense than a simple flyby is an encounter with a comet in which the velocity of the spacecraft is made equal to that of the comet by means of a rocket impulse. Of course, it is an unrealizable dream to impart an additional impulse of 70 km/sec. Thus, we come back to Jupiter and we recall with regard to Saturn that these planets are capable of placing a spacecraft on a countermotion trajectory around the Sun. Haley's comet, having a period of revolution of 76.029 years, an orbit eccentricity of 0.967, a perihelion distance of 0.587 a.u., and an inclination of 162.21° (the accuracy of these values is uncertain, since comet orbits are subject to severe perturbations), is presently returning from its aphelion, is located beyond the orbit of Neptune, and will pass through perihelion on January 8, 1986 (with a possible uncertainty of several months). It has been stated in a paper by Kruze and Fox in 1966 that, to make use of a Saturn swingby, a launch should be performed in the autumn of 1973 or the autumn of 1974, with a specific energy (this name is frequently given to the quantity v_∞^2) of more than $150 \text{ km}^2/\text{sec}^2$, and the space technology is not ready for this event. A launch via Jupiter requires an energy of more than $180 \text{ km}^2/\text{sec}^2$, but need not be performed until the autumn of 1977 or the autumn of 1978. This would give time for consideration of the mission.

It is interesting to consider also an active Jupiter swingby with application of an impulse, as usual, at the pericenter of the swingby hyperbola. Figure 7 (from a paper by Kruze and Fox) shows a trajectory leading to encounter with Haley's comet. We present the flight data (the figures in brackets refer to a similar operation with a passive Jupiter swingby): launch on September 2 (September

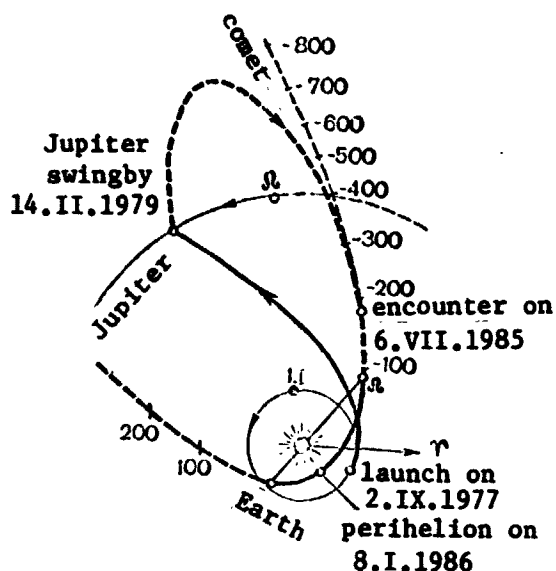


Figure 7. Encounter with Halley's comet during an active swingby of Jupiter. The dotted line shows the part of the orbit lying south of the plane of the ecliptic. The numbers on the comet orbit indicate the number of days before and after perihelion passage.

5.36 (5.83) km/sec; the total characteristic velocity is 23.08 (24.57) km/sec; the useful payload (instruments and control system) reaching the comet is 45 (210) kg. In the active swingby case, the Saturn V launch vehicle imparts the required velocity at emergence from the sphere of influence to a spacecraft of mass 7260 kg; the Saturn V contains four chemical rocket stages (specific impulse 350 seconds): two for the swingby maneuver, and two at encounter with the comet. In the case of the passive swingby, the same rocket places the spacecraft into orbit, and contains only four stages: two for the departure from orbit, and two for the impulse at encounter with the comet.

It is interesting to note that a larger payload can be sent to Halley's comet if one uses an onboard nuclear electric rocket. Its low thrust would initially accelerate the spacecraft and then decelerate it, at the same time transferring the motion into a different plane with opposite rotation, and then would accelerate

13), 1977, with approximate characteristic velocity of departure to an orbit height of 200 km, equal to 9.00 km/sec; the speed of departure from orbit is 7.43 (9.74) km/sec, which corresponds to a speed of departure from the sphere of influence of the Earth of 10.50 (13.60) km/sec; Jupiter swingby on February 14, 1979 (September 16, 1978) at a distance of 20.56 (7.87) radii from the center of Jupiter; the rocket impulse at swingby is 6.29 (0.00) km/sec; the encounter on July 6 (May 27), 1985, at a distance 3.04 (3.79) a.u. from the Sun, at 185 (254) days before the simultaneous passage through aphelion with a rocket accelerating impulse of

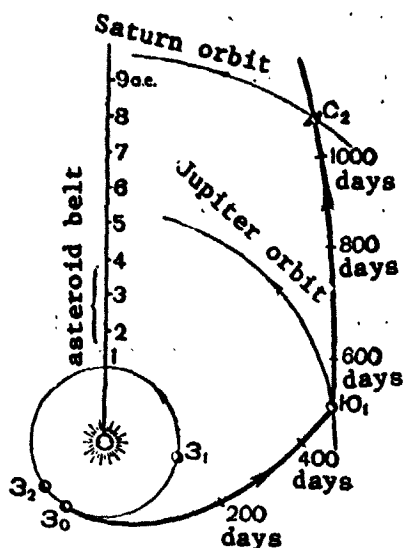


Figure 8. Flight to Saturn via Jupiter. Positions of the planets: E_0 -Earth at launch; E_1, J_1 - Earth and Jupiter at Jupiter swingby; E_2, S_2 - Earth and Saturn at time of arrival at Saturn.

a special role, since, firstly, its mass is very large and, secondly, entry into its sphere of influence occurs with the greatest velocity. The Saturn rings are a prohibiting factor, preventing flight at distances from the planet which otherwise would be particularly suitable in certain cases.

We shall consider several typical missions.

Earth-Jupiter-Saturn (Figure 8). A launch would occur in September, 1977, with initial ideal velocity of 16.5 km/sec (including losses of 1.22 km/sec). After 500 days, the probe passes at a distance of 4 Jupiter radii from the center of the planet and receives an impulse of 18.7 km/sec as a result of the gravitational attraction (the maximum impulse at this distance is 21.3 km/sec). Figure 8 shows a plan view of the heliocentric trajectory near Jupiter. At 1072 days from launch, a spacecraft reaches Saturn,

*These are summarized quite fully in the book by Ts. V. Solov'yev and E. V. Tarasov, *Prediction of Interplanetary Flight* (in Russian), Mashinostroyeniye, Moscow, 1973.

it in the direction towards the Sun, so that the speed of the spacecraft and the comet became equal before perihelion was reached. A launch to this comet would be possible in 1984, when it is hoped that space technology will be adequate. But this falls outside the scope of the present monograph.

Multiplanet Flights

Many investigations of recent years* have shown that many operations will become possible before the end of this century, in which the gravitational fields of planets are used to greatly reduce flight duration. Here Jupiter will play

entering its sphere of influence with a planetocentric velocity of 17.8 km/sec. The initial velocity at departure from Earth is equal to the minimum velocity for direct flight to Saturn, but the swingby around Jupiter has reduced the flight time by a factor of two.

Calculations show that Saturn can be reached even with a velocity equal to the minimum velocity for reaching Jupiter. Flights to Saturn via Jupiter will be possible each year from 1976 to 1979 at opportunities of a month's duration, and the flight trajectory in 1979 will be the least sensitive to initial errors.

The favorable configuration of Earth, Jupiter, and Saturn will repeat again only after 20 years.

The American MJS-77 mission (Mariner-Jupiter-Saturn 1977) calls for launch, using the Titan-3E-Centaur launch vehicle, of two spacecraft of mass 679.5 kilograms at an interval of 20 days. The spacecraft will swing by Jupiter in 1979, and Saturn in 1981. At 19 hours and 1 minute before the Saturn swingby, the spacecraft will pass at a distance of 10,970 km from Titan, which is then moving around Saturn in an orbit of radius 1.22 million kilometers.

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Earth-Jupiter-Saturn-Earth. Favorable opportunities for this mission will occur annually from 1977 to 1983, and then from 1996 to 1999 (the opportunities are separated by the synodic period of Jupiter, 399 days). Here the entry speed into the Earth atmosphere is less than 20 km/sec in all cases (at a perigee height of 111 km), and the launch energy is less than $130 \text{ km}^2/\text{sec}^2$ (i.e., the geocentric velocity v is less than 11.4 km/sec), excluding the flight beginning in February, 1983, where it is $146.7 \text{ km}^2/\text{sec}^2$ ($v_\infty = 12.11 \text{ km/sec}$, which is close to the value corresponding to the third cosmic velocity). If the Saturn swingby trajectory passes through a corridor of width 12,000 km between its rings and the surface of the planet, the duration of the whole mission is less than for a trajectory passing outside the rings, for a launch at the same opportunity. For a launch in 1979, the difference is $3299.8 - 2527.2 = 772.6$ days, i.e., more than two years. The minimum duration

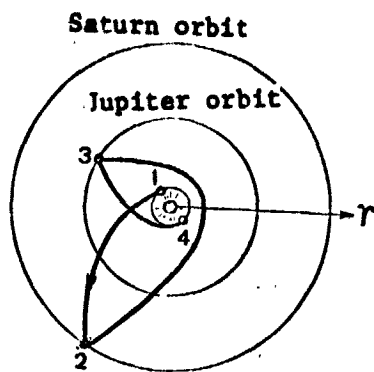


Figure 9. Return to Earth following swingby of Saturn and Jupiter.

1- launch on December 16, 1981;
2- Saturn swingby on January 23, 1986; 3- Jupiter swingby on December 14, 1991; 4- return to Earth on September 1, 1993.

(launch on July 8, 1999) is 2148.2 days (about 6 years), and the maximum (launch on January 8, 1989) is 3654.0 days (ten years).

Earth-Saturn-Jupiter-Earth.

Favorable opportunities, separated by the synodic period of Saturn (378 days), occur from 1979 to 1984 and from 1997 to 1999. All the trajectories, apart from the launch in October, 1979, with a Saturn swingby between the rings and the planet, require a launch energy less than $130 \text{ km}^2/\text{sec}^2$.

The flight durations are:

maximum (launch on December 27, 1982) of 4303.9 days (about twelve years), and minimum (launch on June 14, 1997), and flight between the rings and the planet — 3831.4 days (10.5 years). Figure 9 shows a trajectory corresponding to a launch energy of $125.4 \text{ km}^2/\text{sec}^2$, which swings by Saturn at a distance of 3.26 and Jupiter at a distance of 1.38 of the respective planet radii from the planet center. We note that astronomers are not certain that the outer Saturn ring does not extend beyond the edge as seen from Earth with radius 2.3 equatorial radii of the planet.

Earth-Jupiter-Uranus. Flights will be possible each year from 1978 to 1980. The configuration of the planets relative to the Sun repeats every 14 years. With a velocity of 7.9 km/sec at departure from a 200 km orbit, the flight to Uranus takes 5.04 years.

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Earth-Jupiter-Neptune. Flights are possible from 1977 to 1985. The configuration repeats every 13 years. For a velocity of departure from orbit of 8.2 km/sec, the flight takes 7.56 years. For the same initial velocity, a flight in a Hohmann trajectory takes about 31 years.

Flight trajectories to Saturn, or Uranus, or Neptune via Jupiter are less sensitive to initial errors during the 1979 opportunity.

Earth-Jupiter-Pluto. The configuration repeats every 12 years. Pluto is reached after 8.93 years for a velocity at departure from orbit of 9.0 km/sec. Launches are possible in 1977 and 1978.

Earth-Saturn-Uranus. Flights are possible in 1979 to 1985.

Earth-Saturn-Neptune. Flights are possible in 1977 to 1985.

A swingby of any planet greatly increases the planetocentric velocity at entry into the sphere of influence of the target planet, in comparison with the velocity of entry in a direct flight to the planet. This increases the prospects for using the target planet as a new intermediate planet for a flight to the next. The question can be asked whether one should attempt a series of successive gravitational impulses to accelerate a spacecraft from planet to planet, somewhat after the fashion of a billiard ball.

Earth-Jupiter-Saturn-Uranus-Neptune. Such a flight has been called the "grand tour" in the literature. Figure 10 shows five grand tour trajectories, carried out annually in the period mentioned /48 (during roughly a three-month launch window). These trajectories correspond to Figure 8 in the initial path, in the Earth-Jupiter-Saturn section. The next flight of this kind could begin only in 2154. The beginning and end of the 1976 — 1980 period are determined by the mutual position of Jupiter and Saturn. A launch on September 14, 1977, with energy $120 \text{ km}^2/\text{sec}^2$ reaches Neptune in 9.2 years: Jupiter swingby occurs on April 20, 1979, at a distance of 4.0 planet radii from the planet center, and Saturn swingby occurs on September 3, 1980, at a distance of 1.1 radii (i.e., inside the ring); swingby of Uranus occurs on February 1, 1984, at a distance of 1.9 planet radii from the planet center; and swingby of Neptune occurs on November 8, 1986. The flight to Neptune takes longer (roughly 11 years), if the Saturn swingby path is outside the rings.

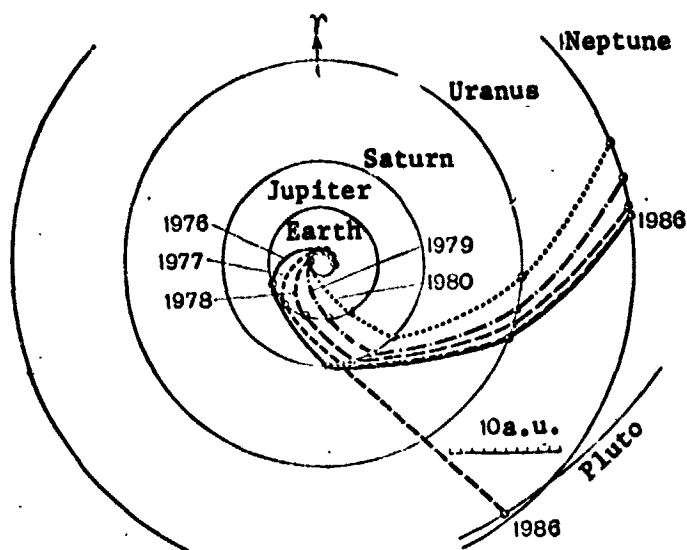


Figure 10. Grand tour Earth-Jupiter-Saturn-Uranus-Neptune trajectories with launches in the period 1976 to 1980 and an Earth-Jupiter-Saturn-Pluto trajectory with launches in 1977.

In all cases, the heliocentric parts, Saturn-Uranus and Uranus-Neptune are considerably hyperbolic. After the Neptune swingby, the heliocentric velocity is several times greater than the local parabolic relative to the Sun, and the space probe aims towards the edge of the Solar System. /49

Simplified variations of the grand tour mission are possible.

Earth-Saturn-Uranus-Neptune. Flights are possible during several years after 1980. These require an initial velocity greater than the third cosmic velocity (launch energy of more than $150 \text{ km}^2/\text{sec}^2$).

Earth-Jupiter-Uranus-Neptune. Favorable opportunities occur from 1978 to 1980. The next period of this type will be 2155 and 2156. For a launch on November 6, 1979, with energy $120 \text{ km}^2/\text{sec}^2$, Neptune is reached after 9.1 years, on November 28, 1988.

Earth-Jupiter-Saturn-Pluto. Favorable opportunities occur in 1977 and 1978. The next such period will be 2076 and 2077. The gravitational maneuver after Saturn must now ensure a different direction for the heliocentric velocity at departure from the Saturn

sphere of influence. For a launch of September 4, 1977, with energy $120 \text{ km}^2/\text{sec}^2$, Pluto is reached after 8.5 years on March 9, 1986, of which period 5.5 years is spent on the almost straight-line Saturn-Pluto section (see Figure 10) (on a parabolic flight, this section takes 17 years).

For various reasons (mainly because of lack of the necessary means), the USA has decided not to pursue the grand tour mission, neither in its entirety nor in the abbreviated variations, but has replaced it by the MJS-77 mission which was discussed at the beginning of this section.

Multi-Planet Flights Using Low Thrust Engines

It has been proposed (Flandreau, USA) to achieve a considerable increase in the payload of a spacecraft sent on a grand tour mission by accelerating it at the beginning of the heliocentric motion by means of a solar-electric rocket engine. Here a payload gain is achieved only if the spacecraft is not aimed directly towards Jupiter (roughly along an impulsive trajectory), but describes a loop around the Sun, which increases the acceleration. The acceleration program is chosen so that entry into the sphere of influence of Jupiter occurs with the maximum possible planetocentric velocity at the maximum payload. To achieve this objective, the flight to Jupiter is extended in time. This time penalty is compensated for later, since the large velocity at entry to the Jupiter sphere of influence leads to a large velocity increment as a result of the gravitational shock, and the Jupiter-Saturn section of the trajectory will be hyperbolic. /50

Figure 11 shows a grand tour trajectory following this approach /51 and beginning in October, 1977 (in that year, the conditions are most favorable for use of a solar-electric propulsion engine). Launch from the Earth sphere of influence is accomplished with an Atlas-Centaur launch vehicle. The solar electric unit operates subsequently for 640 days and is switched on at a distance of 3 a.u. from the Sun. The thrust direction is varied in a complex manner. Neptune is reached 9.4 years after launch.

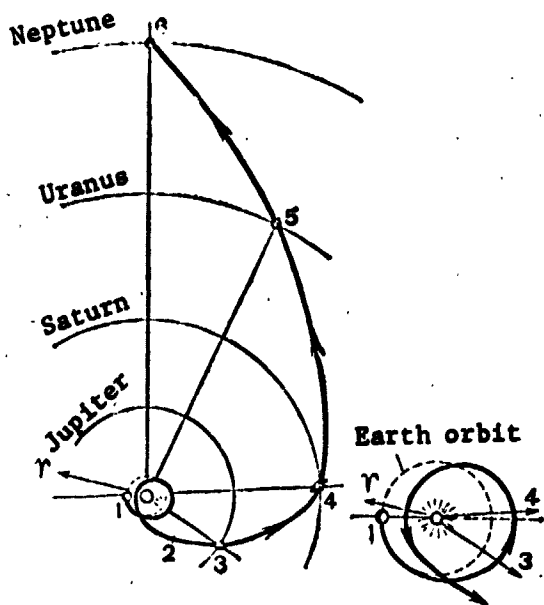


Figure 11. An Earth-Jupiter-Saturn-Uranus-Neptune Grand Tour using a solar-electric engine. The initial part of the trajectory is shown separately on a magnified scale:

- 1- launch (October, 1977); 2- engine is switched on; 3- Jupiter swingby (March, 1980); 4- Saturn swingby (July, 1984); 5- Uranus swingby (December, 1984); 6- Neptune swingby (September, 1987).

There is a wider range of launch date in each year than offered by the purely impulsive grand tour mission, which extends for several months but the flight duration to Neptune and the payload depend markedly on the launch vehicle. Launches in 1976 and 1978 yield a payload 15% less for the same flight duration.

If the objective is not definitely to swing by all four planets, an increase in payload can be achieved. For the Earth-Jupiter-Uranus and the Earth-Jupiter-Neptune itineraries, it is better to launch in September to October, 1978. In July — August, 1976, it is better to fly with a solar-electric engine on the Earth-Jupiter-Pluto itinerary.

In practice, the above flight will apparently not be accomplished in the current century, but work in the USA is reported to equip a modified Pioneer spacecraft with mercury-ion engines, supplied with electrical energy from solar cells at distances out to 5 a.u. from the Sun, and from radio-isotope thermo-electric generators out to greater distances from the Sun. Equipment of this type must facilitate direct flight to planets of the Jupiter group, as well as flight with an intermediate swingby of Jupiter on the way to the more distant planets, or to the Sun, or on a path departing from the plane of the Ecliptic.

One should not forget that, if swingby flights (all of which cannot always be done) are excluded, it is practically impossible to reach the distant planets (with an acceptable mission duration) without using electric or nuclear engines.

CHAPTER 4

FLIGHT TO THE REGIONS BEYOND THE PLANETS

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Direct Flight and Flight Via Jupiter

The objectives of space missions which we have examined up to now were located no further from us than the orbit of Pluto. Now the objective of investigation will be the region of the Solar System beyond the planets (more accurately, beyond Pluto). For brevity, we shall designate this region by the letters BP.

We shall consider the inner boundary of the region beyond the planets (BP region) to be a circle of radius 40 a.u. (the mean distance of Pluto from the Sun). Of course, this is arbitrary: there may be other planets, and even a complete band of asteroids beyond Pluto.

The outer boundary is even more arbitrary. We shall write down the possible factors which enter into its definition.

A sphere of radius 230,000 a.u. is the region of stable direct revolution around the Sun (so the so-called Hill sphere with respect to the Galactic core).*

*The closest star Proxima Centauri is located at a distance of 270,000 a.u.

The sphere of radius 100,000 a.u. is the same thing for revolution in the opposite direction. The comet cloud Oort has a value from 100,000 to 150,000 a.u.

This is the location of the aphelion of the long-period comets, whose perturbations by the stars, as proposed by many (but not all) astronomers, "merge" with circular orbits within the Solar System. When such a comet passes by chance near a planet of the Jupiter group, it can transfer to the orbit of a short-period comet.

The sphere of influence of the Sun with respect to the core of the Galaxy is 60,000 a.u. = 1500 mean distances of Pluto from the Sun = about one light year. /53

The solar magnetospheric drift, which is a perturbation by the Sun of the interstellar magnetic field, extends, it has been postulated, to a distance of 400 to 600 a.u. (10 to 15 mean distances of Pluto). However, the transitional region between near-solar and inter-stellar space extends much further.

The well-known American scientist K. Eyricke has proposed arbitrarily to assume the outer boundary of the BP region to be a sphere of radius 0.1 light year = 6320 a.u.

Actual penetration into the BP region does not present any difficulty. The third cosmic velocity is sufficient. However, to fly in this way to the outer boundary of the BP region requires 35,000 years, and a flight to the boundary of the sphere of influence of the Sun (60,000 a.u.) would require a million years.

Flights to the BP region must, therefore, be hyperbolic relative to the Sun. Each hyperbolic trajectory can be described by its excess hyperbolic velocity V_{∞} , or, if suitable, by the heliocentric velocity of escape from the sphere of influence of the Sun.

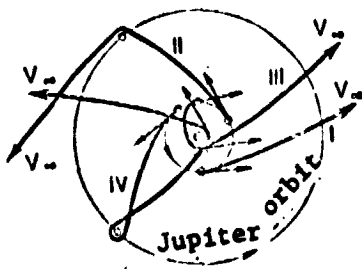


Figure 12. Diagram of different methods for escape to the region beyond the planets.

I- direct flight; II- flight via Jupiter; III- active maneuver near the Sun; IV- active maneuver near the Sun following Jupiter swingby.

If $V_{\infty} = 14$ km/sec, Pluto can be reached in 14 years (instead of 20 years for a flight with the third cosmic velocity). To do this, the departure velocity from a low Earth orbit must be $\Delta v = 13.4$ km/sec.

Earth-Jupiter-BP. We know that efficient acceleration can be obtained by a swingby of Jupiter (trajectory II of Figure 12). For a flight with the third cosmic velocity, this method can give $V_{\infty} =$

24.4 km/sec in the best case. Then a spacecraft reaches Pluto (40 a.u.) in nine years, and reaches a distance of 280 a.u. in 25 years.

The maneuver of maximum efficiency has a departure velocity from low Earth orbit of 18.3 km/sec. Then $V_{\infty} = 46.9$ km/sec, and the spacecraft reaches 600 a.u. in 50 years, while Pluto is reached in three years.

In presenting these data in an article published in 1972 in the *Journal of the British Interplanetary Society*, K. Eyricke proposed several original ideas for accelerated flight. These are discussed below.

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An Active Maneuver Near the Sun

We assume that our objective is to place a spacecraft in a hyperbolic geocentric trajectory with a given value of V_{∞} . Then, we show that, if this value is greater than the local parabolic velocity relative to the Sun in the orbit of the Earth ($V_{\infty} > 42.122$ km/sec), then instead of a direct escape (single impulse) into a hyperbolic trajectory, it is more expeditious to use a two-impulse maneuver; the first impulse directs a spacecraft along a semi-ellipse towards the Sun, and then at perihelion an accelerating

impulse transfers it to the required hyperbola (trajectory III of Figure 12). The total of the two impulses will be less than the one impulse in the direct transfer case, and the gain is thus greater than for a close approach to the Sun. This is well known from applied celestial mechanics.

For example, the velocity of the spacecraft at perihelion before it is given an accelerating impulse, for a distance of 0.1 a.u., is 127 km/sec, for a distance 0.05 a.u., it is 184 km/sec, and for a distance 0.01 a.u., it is 419.1 km/sec. (These velocities are very close to the local parabolic values. For example, for a distance 0.01 a.u., $V_p = 421$ km/sec.) We now evaluate the accelerating impulse. If $V_\infty = 100$ km/sec, then the velocity at perihelion of 0.01 a.u. = 1,496,000 km, after the acceleration must be

$$\sqrt{\frac{2 \cdot 132718 \cdot 10^6}{1496 \cdot 10^6} + 100^2} = \sqrt{187000} = 432.9 \text{ km/sec.}$$

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(The number in the numerator of the fraction under the first square root sign is twice the gravitational parameter K of the Sun.) Thus, the accelerating impulse is $\Delta V = 432.9 - 419.1 = 13.8$ km/sec. Together with the impulse at escape from a low Earth orbit of $\Delta v = 20.07$ km/sec, this gives a total characteristic velocity of 34 km/sec. In fifty years, a distance of 0.017 light years = 1074 a.u. is reached. However, if we wish to reach a distance of 0.1 light year (6320 a.u.) in 50 years, i.e., to depart on a trajectory with $V_\infty = 600$ km/sec, then the impulse at perihelion is 314 km/sec, and the total characteristic velocity (at departure from a low orbit around the Earth) will be 334 km/sec.

If we assume that the spacecraft is a ten-stage rocket system, using engines with gaseous nuclear reactors, offering an exhaust speed of 50 km/sec (fantastic for the present time, but a theoretically possible value), then with a structural characteristic $s = 20$, the relative payload P, from the formula near the end of Chapter 1 is about 2000. The spacecraft must have a thick heat shield, since it flies at a distance of two solar radii from the

center of the Sun,* high-power energy sources, and a huge paraboloid antenna. Its mass, without counting the nuclear reactor installation, would probably be not less than 5 tons. In this case, the launch mass at departure from orbit is roughly 10,000 tons, i.e., it would take roughly 100 Saturn-V launch vehicles, or 400 flights of the Space Shuttle to assemble the entire system. Also, we have proposed the use of gaseous nuclear engines, although there are still many difficulties to be overcome before these become viable.

However, the state of affairs would be improved if, instead of a two-impulse maneuver, the escape along a hyperbolic trajectory were accomplished with a single impulse, departure from the Solar System with a $V_{\infty} = 600$ km/sec. In this case, the impulse at departure from a low Earth orbit is 564 km/sec. That is, an excess of 230 km/sec! In fact, the absence of a heat shield would now reduce the payload mass.

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Maneuvers Near the Sun with Jupiter and Saturn Swingby

It is desirable to search for means of transfer to the region near the Sun with velocities considerably larger than from Earth orbit.

Earth-Jupiter-Sun-BP Region. This variation (trajectory IV on Figure 12) increases the flight time by 3 to 4 years, but scarcely reduces the impulse near the Sun. In fact, the trajectory along which the probe proceeds to the Sun following Jupiter swingby is close to a parabola near the Sun (like the orbits of the short-period comets). However, the difference between the velocity at arrival at perihelion and the local parabolic velocity is very small also in the Earth-Sun-BP Region, and the difference between impulses at departure from Earth orbit for direct flight to the Sun and with a flight to the Sun via Jupiter (for a distance of 0.01 a.u., it is roughly 13 km/sec) is not significant, measured against the huge impulse at perihelion.

*The spacecraft is located at a distance of less than 0.3 a.u. from the Sun for not less than 24 hours.

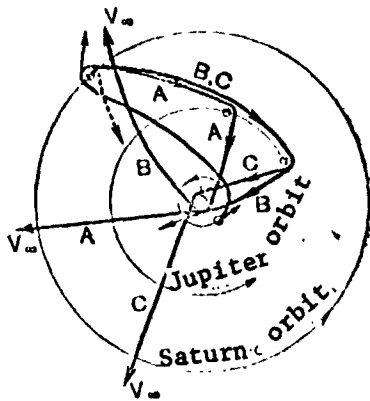


Figure 13. Scheme for several possible variants for the Earth-Saturn-Jupiter-Sun-BP Region maneuver.

Earth-Saturn-Jupiter-Sun-BP Region. For this variation, we have in mind an active maneuver near Saturn, which should transfer the spacecraft into a trajectory opposite in direction to the motion of Jupiter, with perihelion located in the asteroid belt. An impulse at the pericenter does not allow the probe to execute a figure-of-eight flyby (dotted line in Figure 13), and acquire a new direction.

For values of V_{∞} less than about 180 km/sec, no active maneuver near the Sun is required, since the probe departs on the required heliocentric hyperbola following Jupiter swingby. Then the sum of the impulses at departure from Earth orbit and near Saturn will not be greater than about 13 km/sec. (For velocity $V_{\infty} \approx 180$ km/sec, a distance of 500 a.u. is traveled in 13 years, and a distance of 1900 a.u. in 50 years.) This avoids the need for a flight close to the Sun and eases the matter of thermal protection.

However, departure on a hyperbola with $V_{\infty} = 600$ km/sec requires 57 a total characteristic velocity of 270 km/sec, when the impulse near the Sun is imparted at a distance of 0.01 a.u. This confronts space technology with a problem that is insoluble today, and probably even in the future, like the problem of the Earth-Sun-BP Region flight.

Although the same configurations for Earth, Jupiter, and Saturn relative to the Sun repeat, as we know, only every 19.9 years, a period favorable for flights of the type Earth-Saturn (active swingby) -Jupiter-Sun-BP Region arrives roughly every 10 years and lasts about 10 years, with an annual launch window (or roughly 12.5 months). The reason is that different variations are allowable for the position of Jupiter in its orbit, the

directions of the flight around the Sun (Figure 13) and the inclination of the trajectory to the plane of the Ecliptic.

Thus, investigation of the BP Region at a radius in excess of about 50 times the radius of Pluto's orbit is possible even using thermochemical engines, if we consider a 50-year flight to this distance to be acceptable. It is clear that investigation of more distant regions requires the use of nuclear engines with liquid or gaseous cores. We have pointed out only that the complex maneuver Earth-Saturn-Jupiter-Sun-BP Region must play the same kind of important role for investigation of the region beyond Pluto as Jupiter swingby plays for investigation within the planetary system.

CHAPTER 5

INVESTIGATION OF THE DISTANT PLANETS

Missions to the Distant Planets

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It is obvious that one cannot speak of manned missions to the surface of giant planets of the Jupiter type, since such a mission as has been mentioned above is very dubious, even for unmanned spacecraft.

We now consider the possibilities available for injecting a spacecraft into a low orbit around a planet and subsequently returning it to Earth, and we assume that the flight out and back occurs along a Hohmann trajectory. In this case, a mission to an orbit around Jupiter would take two years and 276.5 days, including 197.8 days awaiting a favorable time for the return. A mission to an orbit around Saturn would take 5 years and 298 days, and a mission to other planets would take so long that it hardly warrants serious examination.

Here the total characteristic velocity at launch from a low orbit around the Earth, and with complete use of the Earth atmosphere for deceleration and landing, would be: for Jupiter 49 (29) km/sec; for Saturn 35 (22) km/sec. The numbers in brackets are for the case where the descent from orbit around the planet is made by means of aerodynamic deceleration, and not by means of a retro impulse.

We must certainly exclude, as being unrealistic, flights with thermodynamic engines, and we consider the use of nuclear rocket engines with an exhaust speed of $w = 10$ km/sec. Then, for $s = 20$, we find that the relative payload for a four-stage vehicle at departure from Earth orbit will be 25 when the Jovian atmosphere is used for deceleration at injection into orbit. If we assume a payload of 50 tons for the interplanetary spacecraft, then its total mass will be $25 \times 50 = 1250$ tons. This requires not less than 10 launches of modified rockets of Saturn-V type to place such a spacecraft in orbit. And this is true for nuclear rocket engines which do not yet exist, and for aerodynamic deceleration, which has not been used as yet, even for launch of a Martian satellite.

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We assume that we shall be able, at some future time, to perform a descent into the upper layers of the atmospheres of Jupiter and Saturn in such a way that the spacecraft remains buoyant in the atmosphere. The total characteristic velocities for such missions will be: 79 km/sec for Jupiter, and 53 km/sec for Saturn. Only a new and huge step in the development of nuclear rocket engines (which is possible in theory) will allow this kind of mission to be accomplished. With an exhaust speed of $w = 50$ km/sec (gaseous nuclear rocket engine) and with $s = 20$, a single stage spacecraft will have a relative payload of 3, i.e., under the previous assumptions, at departure from a near-Earth orbit, the ship will have a mass of 150 tons. The launch can be accomplished using a single modified rocket of the Saturn-V type.

The use of gaseous nuclear engines, which, according to foreign theoreticians, can give an exhaust speed of up to 70 km/sec, will revolutionize space technology. Missions will become possible which presently seem fantastic. Before then, however, flights will be possible to artificial satellites of the giant planets.

Promising additional possibilities for manned flights will stem from the use of electric rocket engines.

The hope has been repeatedly expressed that, as space technology progresses, it will be possible, as has already been planned, to use hydrogen extracted from ice, mined from the surfaces of some of the moons of the Jupiter group planets, as a working substance for nuclear rocket engines in subsequent missions. There was an article in 1972 in the AIAA Journal concerning the development of a planned mission to Jupiter and its satellites involving six of a crew in a spaceship, equipped with the "Nerva" solid phase nuclear rocket engine, with a specific impulse of 825 seconds and a flowrate of 41.2 kg/sec. The spaceship mass in low Earth orbit is 2650 tons. Hydrogen is extracted in the Jovian atmosphere and is used for flights to the moons Io, Europa, Ganymede, and Callisto, and in intermediate base is created in orbit around Callisto where the tanks are built to hold hydrogen extracted on Jupiter (an electrolytic installation will fill one tank in four days). The engine section will tow the empty tank to Jupiter for filling. On its return, the ship will be injected into an elliptical Earth orbit at a height of 160 to 19,000 km, from which an interorbital transport spacecraft will transfer it to a circular 160 km orbit. The ship can then be used for new missions.

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Investigation of Planets of the Jovian Group

It is not possible to outline in detail here a program for investigation of Jupiter and other similar planets, nor their satellites. Here everything is essentially secret. Our knowledge of the gravitational fields of these bodies is very incomplete. We do not know the exact dimensions of the planets, and we know very little about the oblateness of Uranus and Neptune. Concerning Pluto, nothing is known, except the rather wide range in which its diameter and mass fall. A study is in progress of the magnetosphere and the radiation belts of the planets (only for Jupiter are these known to exist, for sure), the composition, density, and height of their atmospheres, the composition of cloud formations in the atmospheres, as well as certain odd formations such as the Great Red Spot on Jupiter. One can envision attempts to discover volcanic

activity on Jupiter and the other planets, and a search for sources of powerful radio signals on Jupiter. A particular objective of the investigation is the Saturn rings: we know neither their thickness nor the dimensions of the particles composing them.

Why is one hemisphere of Japetus, a moon of Saturn, six times brighter than the other? Why is Io, a moon of Jupiter, orange, and why does it become brighter when it falls in the shadow of Jupiter? We cannot even list all the enigmas.

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The Soviet astronomer I. S. Shkovskiy thinks that one should not completely exclude the possibility of detecting forms of life on Jupiter or Saturn; these forms will differ from Earth types, and will be based on quite different forms of exchange.

The solution of many of these questions has a direct bearing on the construction of spacecraft. Before the first flight to Jupiter, it was assumed that the radiation belts around the planets (particularly around Jupiter) had such a large electron density that they were capable of rendering semiconductor equipment unserviceable. It was considered, therefore, that the orbits of artificial satellites of Jupiter should not have a pericenter closer than 6 Jupiter radii to the center of the planet, although this can be reconciled, apparently, with a single flyby at a closer distance. One may think that the powerful Jovian gravitational field would hold within the sphere of influence of the planet a vast amount of cosmic dust, meteorites, ice particles from the cores of comets of the Jupiter group (comets whose aphelia are close to the orbit of Jupiter), and that one should take into account the meteorite hazard.

Penetration into the upper layers of the Jovian atmosphere should yield information on the denser layers. Scientific questions make it most desirable to enter the atmosphere on the day side of Jupiter, and in one of the American papers, it is recommended that the first probe avoid "special" points on the planet, e.g., the polar regions, equatorial zones, and the Great Red Spot. It is

desirable to transfer the probe radio signals to Earth via a spacecraft bus, moving on a flyby hyperbola. The probe mass could then be 160 kg, and the total mass of the spacecraft on the trajectory to Jupiter would be 520 kg. Separation of the probe from the bus and its injection into a descent trajectory must occur at a distance of 45 millions of kilometers from the planet. A mission of this kind is listed in some of the USA plans, with launches in 1978 or 1980.

The First Flights to Jupiter

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On March 3, 1972, using a three-stage Atlas-Centaur rocket, the spacecraft Pioneer 10, of mass about 250 kg, was launched to Jupiter. The initial velocity was 14.3 km/sec. Trajectory correction maneuvers were carried out on March 7, March 23, and September 13, 1972. In July 1972, the spacecraft passed into the asteroid belt from which it emerged in the middle of April, 1973. In December, 1973, it flew by Jupiter at a minimum distance of 130,000 km from the edge of the atmosphere, with a planetocentric velocity of 36.5 km/sec. Minutes later, it passed at a distance of 18,000 km from the closest moon of Jupiter, Althea, and 17 minutes later, it passed behind the moon Io for 91 seconds, and experienced a radio occultation by its atmosphere. The spacecraft emerged from the sphere of influence roughly in the direction of Jupiter's orbital motion, with a hyperbolic heliocentric velocity of 20 km/sec. The spacecraft is leaving the Solar System with a residual velocity at infinity of $V_{\infty} = 11.4$ km/sec, directed towards the Constellation Taurus. In 1987, it will intersect the orbit of Pluto. Contact with it will probably be maintained until 1979 (until the orbit of Uranus).

Before reaching Jupiter, Pioneer 10 transmitted a great deal of information to Earth. In particular, it verified that the asteroid belt was free of meteorites which would constitute a hazard for spaceflight, but a large amount of dust was detected in the region between the orbits of Earth and Mars ("the great space vampire"). Preliminary results of the investigations of Jupiter radiation during the flyby show that the intensity of the radiation

in three belts of Jupiter is 10,000 larger than for the Earth's radiation belts (and not a million times, as was anticipated by some astronomers). Pioneer 10 entered the radiation belt at a distance of 8 million kilometers from the planet. The night temperature at Jupiter proved to be the same as the day temperature (-133°C), which is due to the fast rotation of the planet. Bombardment by dust particles near Jupiter proved to be 300 times more intense than in interplanetary space, but no dust rings, analogous to the Saturn rings, were observed. The structure of the planet's magnetic field proved to be very complex (it consists of two regions), and the intensity is less than expected, and directed inwards to the south (and not towards the north, as on Earth). The Jovian atmosphere consists mainly of hydrogen, helium, ammonia, and methane, as well as deuterium, acetylene, and ethylene. The spacecraft took 340 color photographs of the planet, as well as photographs of its moons Io, Europa, Ganymede, and Callisto. The surface temperatures of these moons were -173°C . Io has a very rarefied atmosphere of thickness 110 km, and a layer of ionosphere.

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On April 6, 1973, the spacecraft Pioneer 11 was launched, of mass 231 kg and velocity 14.5 km/sec. The fuel available for trajectory correction was calculated to be a characteristic velocity of 200 m/sec. A trajectory correction in April, 1974, adjusts the destination of the spacecraft following Jupiter swingby on December 5, 1974. It will either fly by Jupiter as did Pioneer 10, or, passing Jupiter at a distance of 160,000 km, will be directed to Saturn, or escape from the plane of the Ecliptic, or fly by Jupiter at a distance of 36,000 km from the planet with a velocity of 48 km/sec. The results of the Pioneer 10 flight give reason to believe that a flyby at this distance will not put the scientific equipment out of action.

The USA plans in the next few years provide for the launch of two Mariner spacecraft in the period August — September, 1977, on an Earth-Jupiter-Saturn itinerary, as well as a possible launch, in the period July — August, 1976, of a Pioneer spacecraft to Jupiter where it will carry out a perturbation maneuver and escape from the plane of the Ecliptic.

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